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Technical Memorandum No. 33-114

Direct-Ascent vs Parking-Orbit Trajectory For Lunar-Soft-Landing Missions

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JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

December 3, 1962

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ABSTRACT

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Differences between direct-ascent and parking-orbit modes of transit to the Moon and their effects on a lunar-landing mission are studied within the context and constraints of the Surveyor Project. Constraints considered (at both "current" and "minimum" levels) include lunar lighting, launch-window duration, landing location, launch azimuth, launch vehicle capability, transit time, observability of landing from Earth, launch opportunities per period, and desired mission frequency. A listing of advantages and disadvantages provides some basis for conclusions. Appendices provide source material and amplification, together with a glossary of terms used.

I. INTRODUCTION

The purpose of this study is to discuss some aspects of direct-ascent vs parking-orbit lunar ascent trajectories[1] in the context of the *Atlas/Centaur*-boosted *Surveyor* Project.

The basic objectives of the Project are presented, along with those current system constraints (established to meet these objectives) which relate to this study. An analysis of these constraints is then made, and minimum constraints are set forth accompanied by a discussion of the effect of the constraints. When these minimum constraints are used, direct-ascent trajectories have a reasonable probability of permitting two, or perhaps three, launchings per year without compromising the probability of achieving the basic Project objectives. Parking-orbit trajectories will permit five or six launchings per year even when the more rigorous current constraints are used.

Possible advantages of direct-ascent trajectories, all of which relate to the launch vehicle, are then presented. Spacecraft System aspects relating to the adoption of direct-ascent trajectories are next described. Finally, a summary comparison of the advantages and disadvantages of direct-ascent and parking-orbit trajectories as seen by the Project, the spacecraft, and the launch vehicle, is given.

[1]See Appendix A for explanation of terms.

The basic Project objectives[2] are:

1. To develop a technology for and accomplish a series of soft landings on selected areas on the surface of the Moon.
2. To transport, soft-land, and perform scientific experiments on the surface of the Moon for the purpose of local area investigation.
3. To obtain engineering data which will aid in future space exploration.
4. To telemeter the scientific and engineering data back to Earth for retrieval, reduction, and dissemination.
5. To achieve a 30-day minimum period of operation on the lunar surface for all launches, with a 90-day period of operation as a desired objective.

[2]As set forth in the *Surveyor* Spacecraft System design specification.

II. LAUNCH CALENDAR

The comparison between the direct-ascent and parking-orbit mode of ascent into a lunar transfer orbit is perhaps most striking when the calendar of launch opportunities is determined. As shown in succeeding Figures, the direct-ascent mode is severely limited relative to the parking-orbit mode, which is basically unlimited.

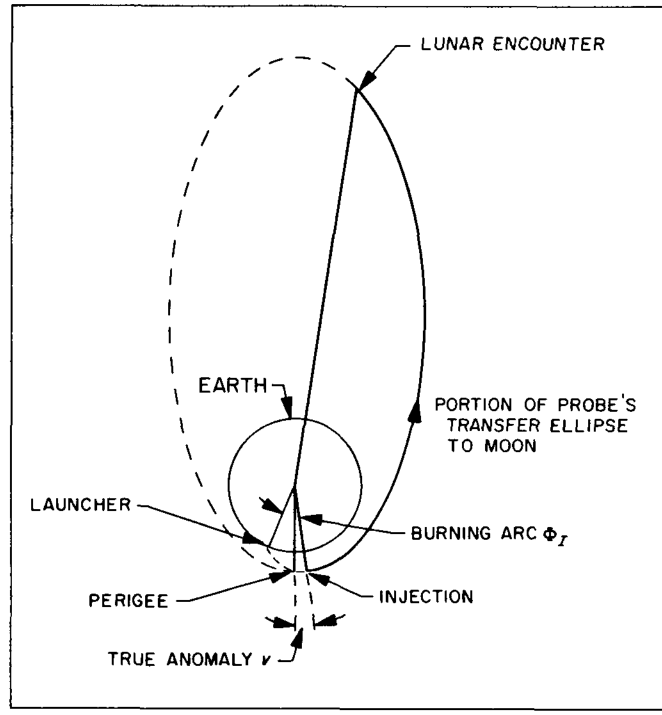
The reason for this difference lies in the very restrictive geometrical configuration of the direct-ascent mode. The situation may be likened to that of a marksman poised in rigid firing stance on a rotating platform and attempting to shoot a slow-flying duck. Obviously, if his movements are very restricted, he can only fire over a small time interval when the platform position relative to the duck brings the latter into his field of view. If, however, he were given considerable freedom of movement, his firing opportunities could be increased significantly. This analogy serves to give a simple illustration of the restrictive properties of the direct-ascent mode and the subsequent relief afforded by using the parking orbit.

A. Mathematical Model for Direct Ascent

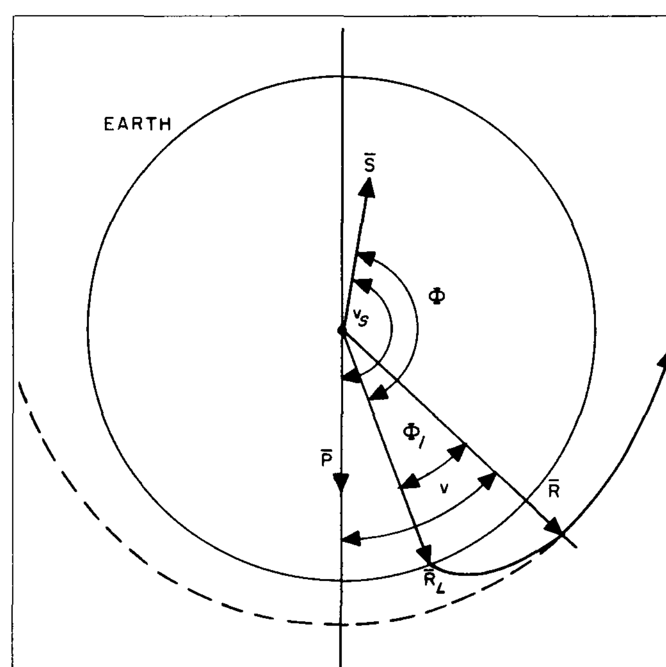
A more rigorous treatment, however, is contained in the mathematical model of the lunar trajectory problem. Figure 1a is a general illustration of the in-plane geometry of the direct-ascent lunar trajectory. The angular relationships for such an orbit are further described in Fig. 1b. Here, **P** is a unit vector in the direction of perigee, $\mathbf{R}_L$ is a vector from Earth's center to the launcher, **R** is a vector from Earth's center to injection in the lunar transfer orbit, **S** is a unit vector in the direction of the Moon at *arrival* time, v is the true anomaly of injection, Φ_I is the burning arc from launch to injection, Φ is the central angle swept out by the vehicle in going from launch to the Moon, and finally v_S is the true anomaly of the Earth-Moon (or **S**) vector in the lunar transfer ellipse.

A full description of the problem requires use of three-dimensional geometry, provided by Fig. 2, which shows the angular relations between launch azimuth Σ_L, declination Φ_L and right ascension Θ_L of the launcher, declination Φ_S and right ascension Θ_S of the Moon at arrival, and the central angle Φ. From this geometry we can write a law of cosines from spherical trigonometry.

$$\cos\Phi = \cos\Phi_L \cos\Phi_S \cos(\Theta_S - \Theta_L) + \sin\Phi_L \sin\Phi_S \quad (1)$$



(a)



(b)

Fig. 1. In-plane geometry, direct-ascent lunar trajectory

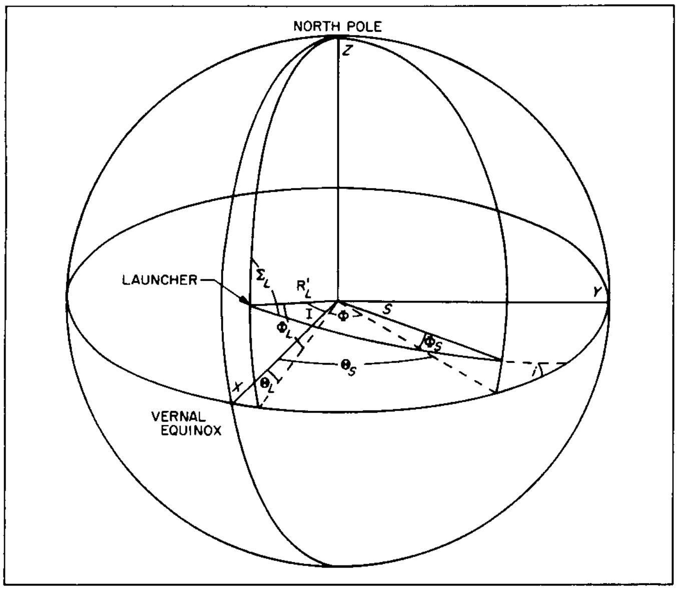


Fig. 2. Direct-ascent launch geometry (three dimensions)

Let $\Theta_S - \Theta_L = \Delta\Theta$ and solve for $\cos \Delta\Theta$, or

$$\cos \Delta\Theta = \frac{\cos \Phi - \sin \Phi_S \sin \Phi_L}{\cos \Phi_L \cos \Phi_S} \tag{2}$$

and from the law of sines we have

$$\sin \Delta\Theta = \frac{\sin \Sigma_L \sin \Phi}{\cos \Phi_S} \qquad \text{for } 0 \leq \Delta\Theta \leq 2\pi \tag{3}$$

and where from another law of cosines

$$\cos \Sigma_L = \frac{\sin \Phi_S - \cos \Phi \sin \Phi_L}{\sin \Phi \cos \Phi_L} \qquad \text{for } 0 \leq \Sigma_L \leq \pi \tag{4}$$

Now the central angle Φ is composed of three quantities (Fig. 1b):

$$\Phi = v_S + \Phi_I - v \tag{5}$$

and for lunar trajectories the true anomaly of the Moon at encounter is relatively constant[3]. For 66-hr trajectories $v_S = 170$ deg and for 90-hr transit the value rises to only $v_S = 175$ deg. Also the burn arc Φ_I is relatively fixed, being about 29 deg for the *Centaur* vehicle though varying about ± 1 degree for different ascent shapes. We shall neglect this variation in this analysis for the present.

Reviewing Equations (1)–(5) it is noted that both the right ascension difference $\Delta\Theta$ and launch azimuth Σ_L depend chiefly on two key quantities, the declination of the Moon Φ_S and the true anomaly v. The launch-site latitude is, of course, fixed at approximately 28.309 deg.

[3]Clarke, V. C. Jr. "Design of Lunar and Interplanetary Ascent Trajectories," Technical Report No. 32-30, Revision 1, Jet Propulsion Laboratory, March 15, 1962.

Assume for the moment that the right ascension Θ_S and declination Φ_S of the Moon are fixed over the launch window. This is a fair assumption since the Moon orbits the Earth at 1/27.3 the Earth's rotational rate. Then a change in the launcher right ascension Θ_L caused by rotation of the Earth must be balanced in Equation (1) by a corresponding change in the central angle Φ. Specifically, the change in Φ must be in the true anomaly v, since the true anomaly v_S of the Moon, and the burn arc Φ_I are relatively invariant.

Thus, it follows that a direct relation exists between launch time T_L (on which Θ_L depends), launch azimuth Σ_L and true anomaly v. Let $\Delta\Theta_1$ and Σ_{L1} be the right ascension difference and launch azimuth corresponding to a launch time T_{L1} and $\Delta\Theta_2$, Σ_{L2} be the same quantities corresponding to a launch time T_{L2}, then the *launch time window* ΔT_L is given by

$$\Delta T_L = T_{L2} - T_{L1} = \frac{\Delta\Theta_2 - \Delta\Theta_1}{\omega} \qquad \text{for } T_{L2} \geq T_{L1} \tag{6}$$

where ω is the earth's rotational rate ($= 4.1780742 \times 10^{-3}$ deg/sec). The launch azimuth interval is subsequently

$$\Delta\Sigma_L = \Sigma_{L2} - \Sigma_{L1} \tag{7}$$

Evaluation of these equations is graphically presented in Fig. 3 for lunar declinations from -17 deg to -29

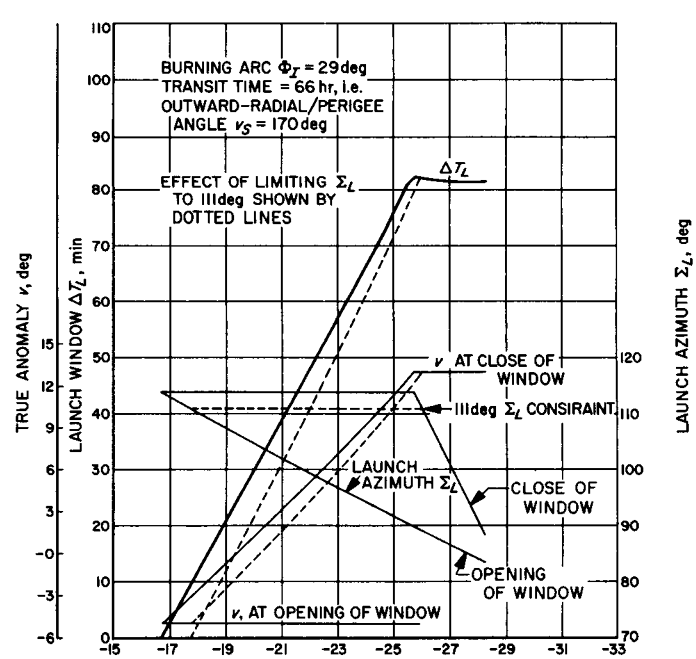


Fig. 3. Launch window, launch azimuth, and true anomaly vs lunar declination

deg for 66-hr trajectories and further illustrated in Appendix B for 90-hr trajectories.

The launch window and launch azimuth are seen to be direct functions of the true anomaly of injection, as shown in Fig. 4, where true anomaly goes from -5 deg at opening to $+6$ deg at close of window, as opposed to the -5 to $+13$-deg range provided in Fig. 3.

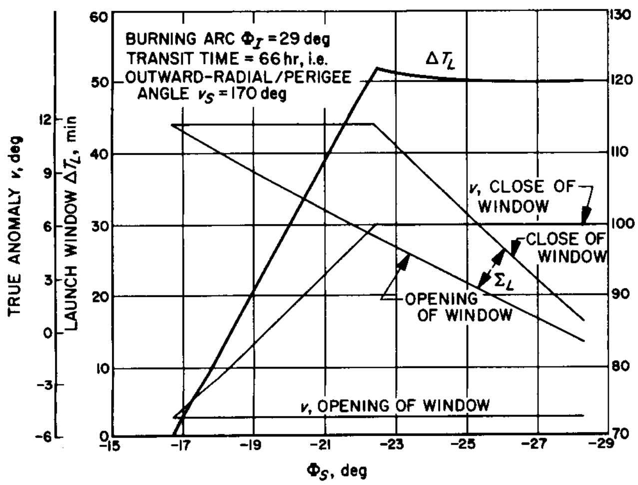


Fig. 4. Launch window, launch azimuth, and true anomaly vs lunar declination for $-5 < \nu < +6$ deg

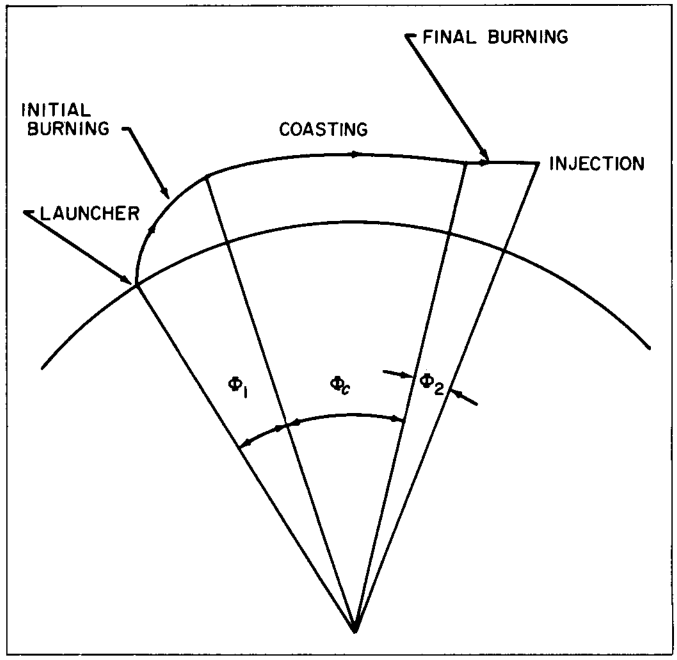


Fig. 5. Parking-orbit burn arc

B. Parking-Orbit Characteristics

Such is not the case for the parking-orbit mode. Here, true anomaly at injection v is held constant. In place of a variation in it, the burn arc Φ_I is modified by splitting it into three parts: (1) a fixed burn arc Φ_1 from launch to parking-orbit injection, (2) a *variable* coast arc Φ_C in the parking orbit, and (3) a final fixed burn arc Φ_2 from parking orbit to final injection as shown in Fig. 5. Theoretically, the coast arc Φ_C can be changed without limit and can provide more than adequate launch window. Practically, however, limitations in launch azimuth, orbital decay effects, duration in parking orbit, tracking coverage, etc., restrict parking-orbit coast time. In spite of these restrictions, the parking-orbit launch windows are considerably larger, being 1 to 4 hr as compared to less than 1½ hr for direct ascent.

C. Direct-Ascent Launch Calendar

Actual launch windows may be extracted from ephemeris data by application of the ΔT_L vs Φ_S relationship. Figure 6 presents a calendar of possible launch dates from July 1964 through September 1966. Both launch and arrival dates are given (66-hr transit assumed), and launch days and windows are shown with and without the requirement of 72 or 150 hr of post-landing sunlight.

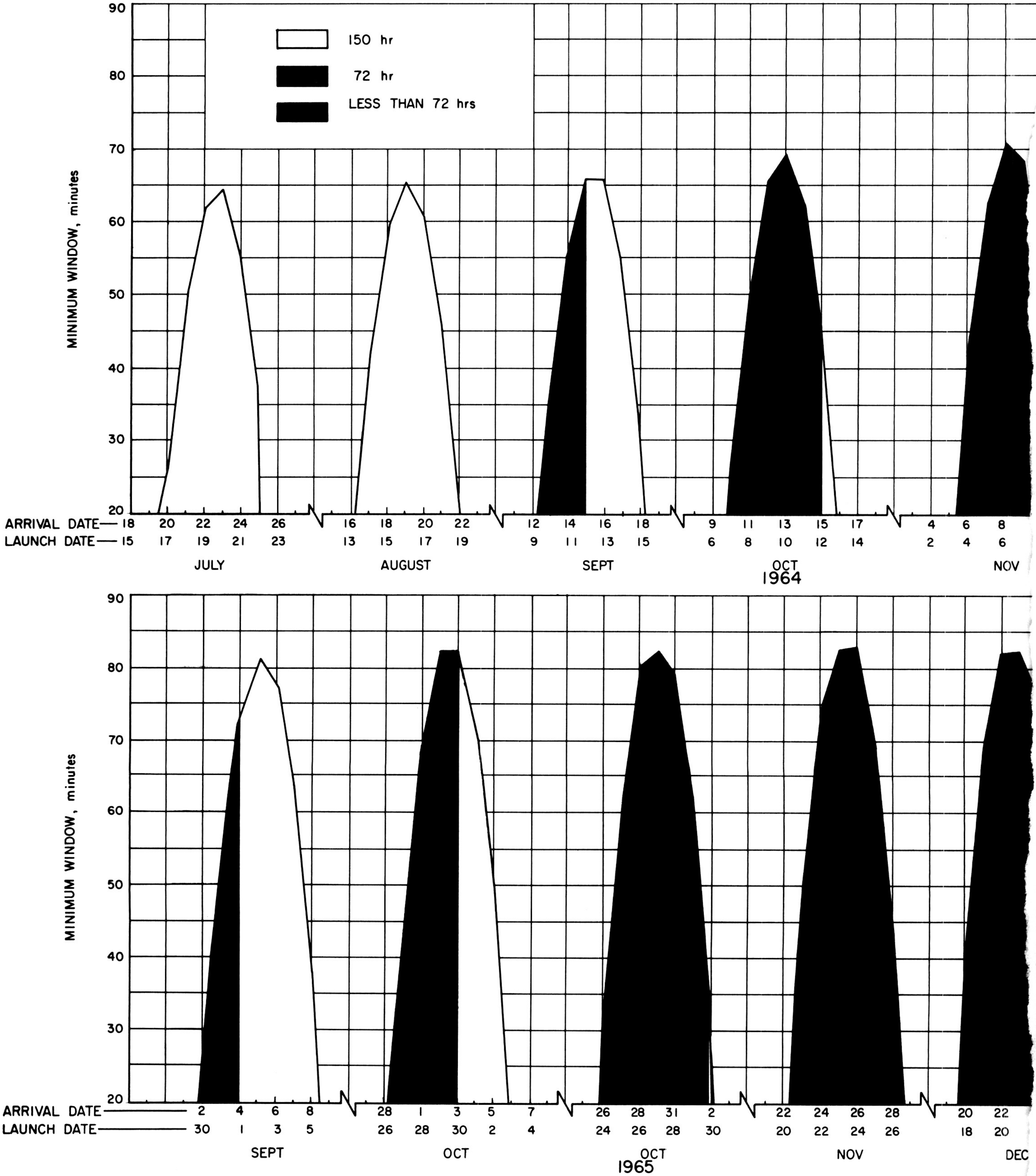
First, to obtain the launch calendar without regard to lighting constraints, the ephemeris is entered with the ΔT_L vs Φ_S curve from Fig. 3.

The post-landing lighting is inserted by consideration of the Sun-Earth-Moon angle (LPH). The landing location is considered to be varied between 8 deg W and 65 deg W longitude to obtain maximum lighting. Within this region, the current requirement of 150 hr minimum post-landing sunlight may be obtained when LPH is between 98 and 258 deg. For the minimum requirement of 72 hr of sunlight, LPH is between 98 and 298 deg. (Pre-landing lighting requirements are neglected). Ephemeris data indicate those days when the LPH falls within these ranges.

The areas under the window-duration curve on the launch calendar are tinted accordingly. Those which are white have at least 150 hr sunlight after landing, the grey indicates 72-hr minimum, and the black have less than 72 hr (down to landing in darkness).

The constraints on true anomaly, launch azimuth, burning arc, and transit time associated with Fig. 3 are equally applied to Fig. 6.

A similar calendar for the 90-hr-transit case is given in Appendix B.


150 hr
72 hr
LESS THAN 72 hrs
MINIMUM WINDOW, minutes
90
80
70
60
50
40
30
20
ARRIVAL DATE— 18 20 22 24 26 16 18 20 22 12 14 16 18 9 11 13 15 17 4 6 8
LAUNCH DATE— 15 17 19 21 23 13 15 17 19 9 11 13 15 6 8 10 12 14 2 4 6
JULY
AUGUST
SEPT
OCT
NOV
1964
MINIMUM WINDOW, minutes
90
80
70
60
50
40
30
20
ARRIVAL DATE 2 4 6 8 28 1 3 5 7 26 28 31 2 22 24 26 28 20 22
LAUNCH DATE 30 1 3 5 26 28 30 2 4 24 26 28 30 20 22 24 26 18 20
SEPT
OCT
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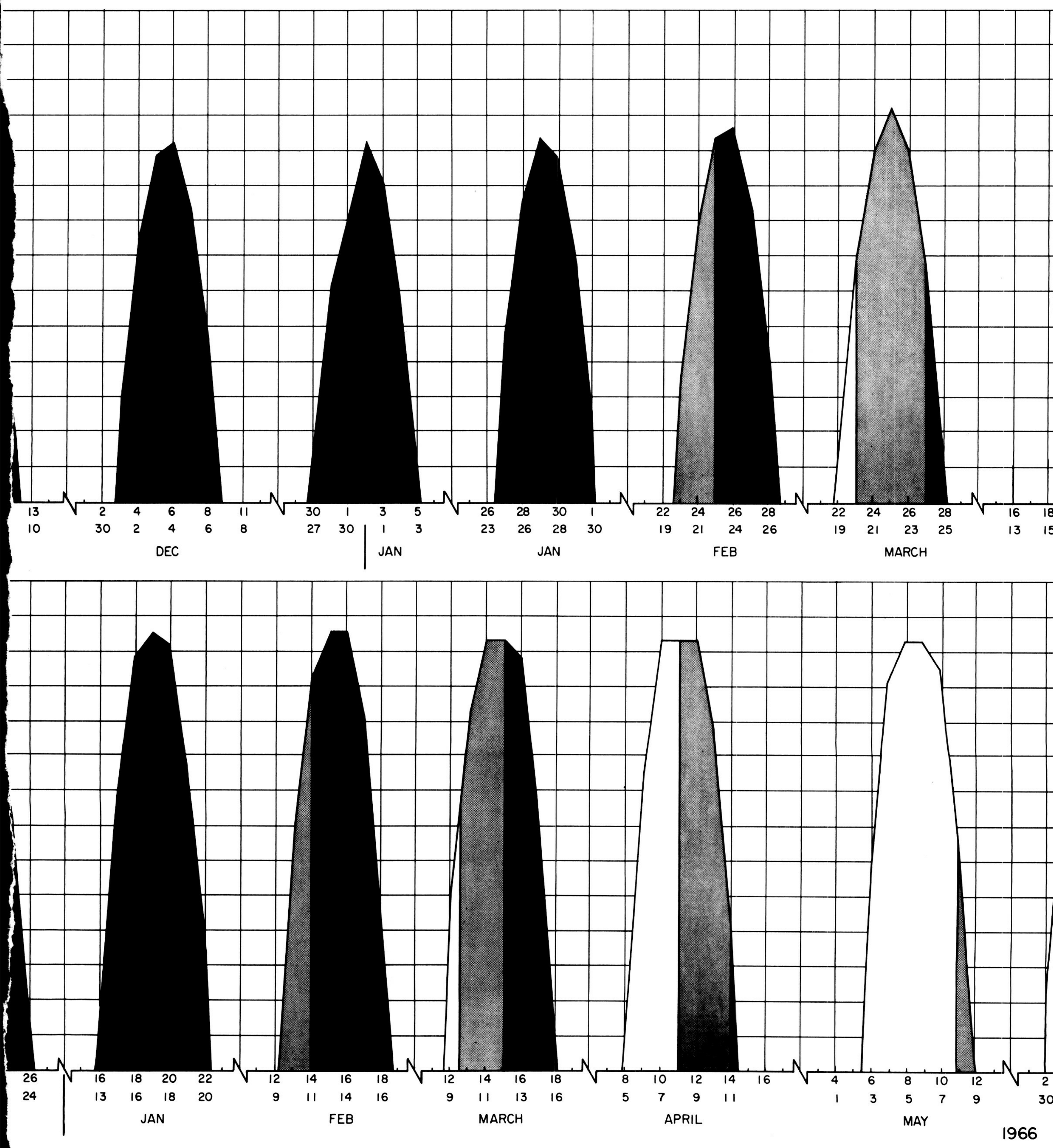
NOV
DEC
1965



13 2 4 6 8 11 30 1 3 5 26 28 30 1 22 24 26 28 22 24 26 28 16 18
10 30 2 4 6 8 27 30 1 3 23 26 28 30 19 21 24 26 19 21 23 25 13 15
DEC
JAN
JAN
FEB
MARCH
26 16 18 20 22 12 14 16 18 12 14 16 18 8 10 12 14 16 4 6 8 10 12 2
24 13 16 18 20 9 11 14 16 9 11 13 16 5 7 9 11 1 3 5 7 9 30
JAN
FEB
MARCH
APRIL
MAY
1966

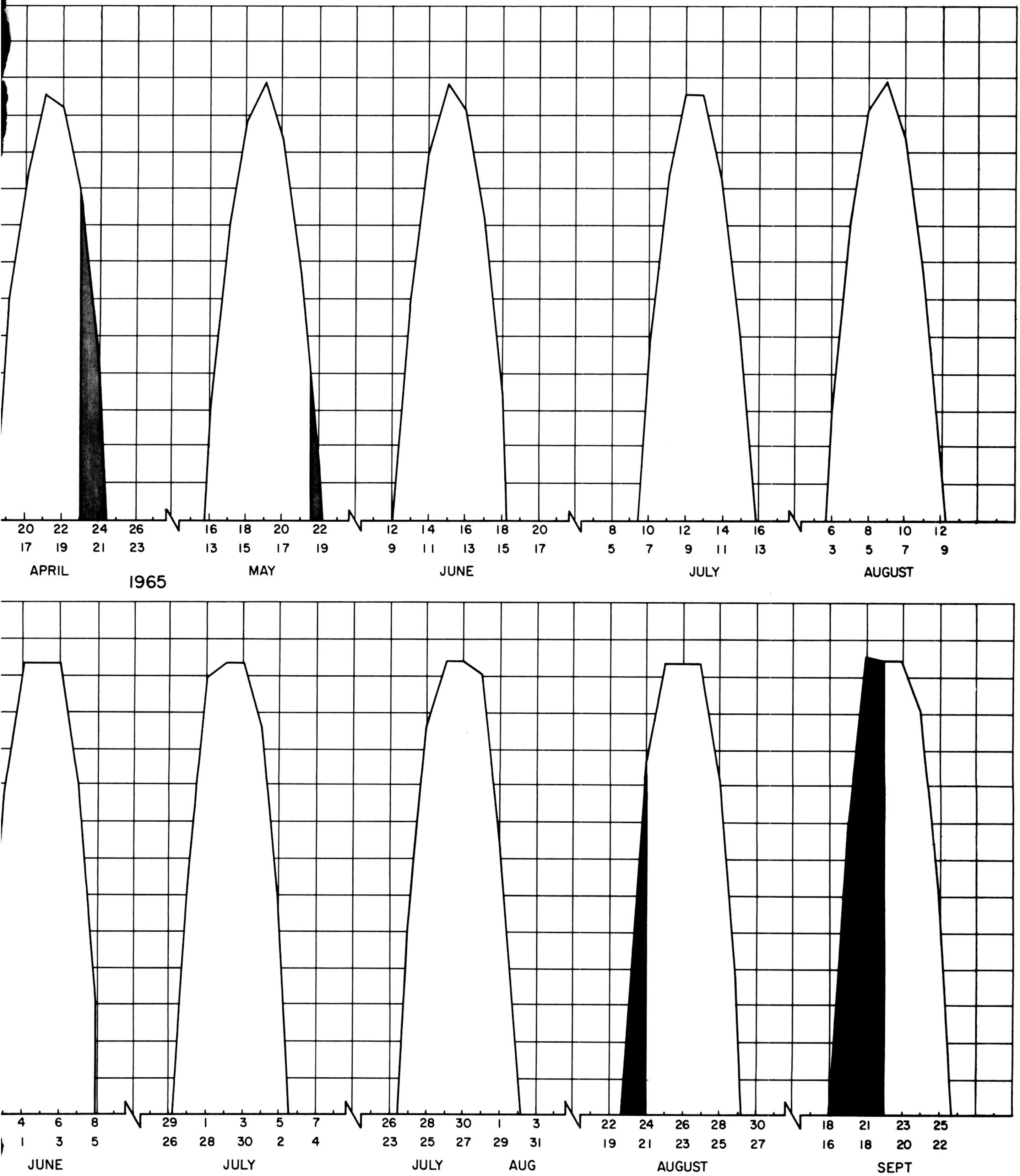


Fig. 6. Direct-ascent launch calendar

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III. CONSTRAINTS AND THEIR EFFECTS

Certain constraints determine the feasibility of using direct-ascent and/or parking-orbit lunar trajectories. They are discussed one by one below according to the following scheme: the constraint as specified for *Surveyor* (without regard to direct ascent) is described, then an estimate is made of the minimum constraint, then the effect of each on the direct-ascent mode is discussed.

The values currently assigned to these constraints in the *Surveyor* governing documents do not necessarily permit direct-ascent trajectories to fulfill the objectives, since they were selected for parking-orbit trajectories. The purpose of this study was to determine feasibility of direct-ascent trajectories when the constraints are modified to be compatible with direct ascent but do not compromise mission objectives.

A. Launch Vehicle Capability

Current requirements specify a parking-orbit capability, a separated payload weight of 2100 pounds minimum, and an RMS injection error not greater than 10.15 meters/sec (1 sigma) 14 hours after injection.

Minimum requirements are assumed to be that the launch vehicle must be able to inject a separated payload weight of at least 2100 lb, when the true anomaly is varied between -5 and $+13$ deg, and the injection shall not have an RMS error greater than 10.15 meters/sec (1 sigma), 14 hours after injection. These requirements are based upon the following:

1. The spacecraft weight allowance must be at least 2100 lb to enable it to carry out its scientific mission.

2. The nominal A-21-configuration midcourse-correction capability is 10.15 meters/sec (1 sigma); the maximum capability (when vernier tanks are full, and vernier fuel is used for ballast to increase the spacecraft main-retro burnout velocity as is currently planned) is 12.7 meters/sec (1 sigma), and would increase spacecraft weight by 13.1 lb. It may be possible, through additional mechanization, to ballast the spacecraft in some other manner. If this could be done, the maximum midcourse correction capability would be 19.3 meters/sec (1 sigma). This would increase the spacecraft weight by about 48.6 lb (see Appendix C).

3. True anomalies between -5 deg and $+13$ deg are considered achievable from a payload and aerodynamic heating point of view. For small values of true anomaly, the launch vehicle payload is reduced about 25 pounds for a ± 1-deg change in true anomaly. Fig. 3 is based upon the -5 deg and $+13$ deg values.

Effects. Increased payload capability could be used to increase both the positive and negative true anomalies and result in increased launch window durations. Figures 3 and 4 demonstrate the effect of changing true anomaly on launch window duration vs lunar declination. For example, increasing the true anomaly from between -5 deg and $+6$ deg to between -5 deg and $+13$ deg increases the maximum launch window for lunar declinations more negative than -22.5 deg to about 80 minutes. Increased payload capability could also be used to increase midcourse correction fuel (up to vernier tank capacity). It could be used to increase the number of scientific experiments, instrumentation, or spacecraft reliability, or to decrease spacecraft cost by permitting fabrication using heavier but less expensive materials.

B. Launch Azimuth

Current requirements specify that the spacecraft shall provide capability for being launched on any azimuth between 90 deg and 114 deg East of true North.

Minimum requirements are between 83 deg and 114 deg East of true North. The difference is based upon the following argument:

1. The 114-deg azimuth is an AMR safety requirement.

2. The 90-deg figure can be reduced without significantly adding to the tracking and telemetry problem.

3. Section II indicates that for the maximum negative lunar declination a launch azimuth of about 83 deg is required at the opening of the window if the maximum launch window is to be achieved.

Effects. Decreasing the 83-deg value would have no effect upon launch window duration as shown in Fig. 3, unless the true anomaly could be made more negative than −5 deg, for which case the launch window duration would be increased about four minutes per degree decrease in true anomaly for lunar declinations more negative than −25 deg. Decreasing the 114 deg would reduce the available launch window about 8 minutes (see Fig. 3). This, in turn, would reduce the number of launch days available for direct-ascent trajectories.

Launch-to-injection trajectories over the 85-to-114-deg launch azimuth ranges are mapped in Fig. 7. Direct-ascent injection points are seen to fall in a narrow band near but not within sight of AMR, while parking-orbit injections, depending on the lunar declination may occur 30 deg South of the Equator and at 90 deg East longitude.

C. *Launch Window Duration*

Current requirements specify that a minimum 50-minute continuous firing window shall be available. The parking-orbit case provides windows up to about four hours in length.

Minimum requirements. The minimum launch window (and minimum number of launch days requirement) cannot be determined on a rigorous basis – rather, it is based upon judgment, past experience, knowledge of all possible launch window durations, spacecraft and launch vehicle technical characteristics, and the number of activities which must be coordinated at the time of launch. In general, the probability of making a successful launch-

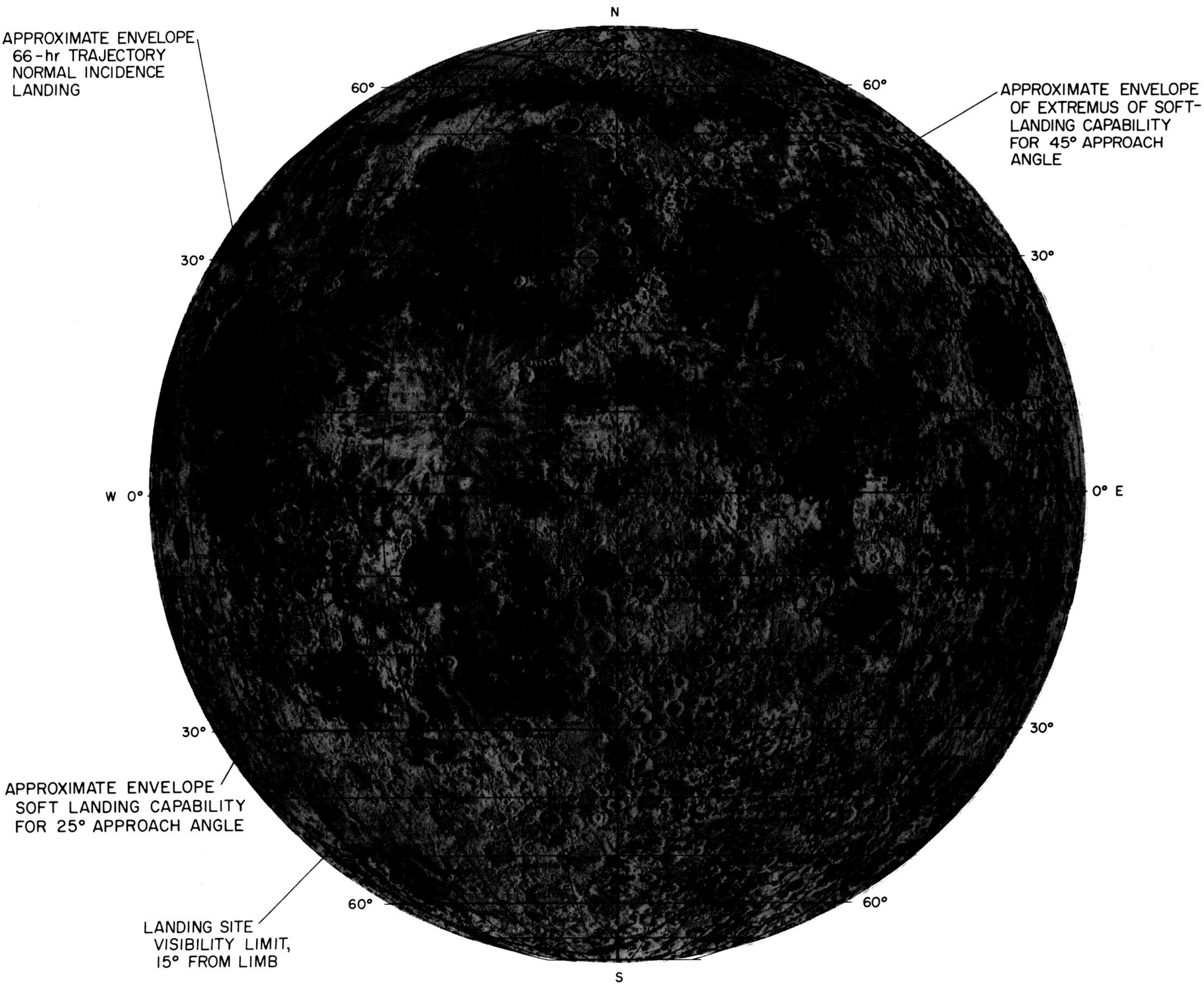

Fig. 7. Typical lunar landing regions

ing should increase as a function of launch window duration and the number of successive launch days available in a launch period. Intuitively, this probability may be expected to increase rapidly to perhaps one hour (launch window) and five successive launch days, and then tend to level off as these values increase further. To get a better grasp on this problem, it is appropriate to consider the launch problems associated with the spacecraft and the launch vehicle separately.

1. Spacecraft limitations. The spacecraft is designed to be checked out and assembled with its shroud (encapsulation) in the AMR explosive safe area. The spacecraft is designed to operate properly and have a launch capability over a ten-day period following encapsulation. (One day for encapsulation, one day for assembly with launch vehicle, and eight days for launch availability). The spacecraft cannot be adjusted or repaired without removing the shroud. Following encapsulation, the spacecraft is transported to the launch pad and mounted on top of the launch vehicle. In this position the spacecraft is monitored and readied for launch through an RF link between the spacecraft and the *Surveyor* Launch Operations Trailer (SLOT). If the spacecraft experiences any difficulties which would prevent launching, it must be removed from the launch vehicle and returned to the explosive safe area for shroud removal and repair. In this event, a second spacecraft will be standing by to be substituted for the defective spacecraft. According to present estimates it will take more than 24 hours and less than 48 hours to replace a spacecraft after T - 8 minutes in the launch countdown. These factors lead to the following conclusions:

Since the spacecraft in itself cannot be adjusted, repaired, or replaced within any reasonable length of launch window, and since the spacecraft replacement time is a little over 24 hours, launchings cannot be made on successive days in case of spacecraft failure, but they could be made on alternate days.

There is a possibility that defective spacecraft ground support equipment could be identified and replaced with spare units within a reasonable launch window. Launch data relating the frequency of GSE failures and repair times are necessary to determine a minimum useful launch window. In the absence of such data, a realistic minimum repair time figure would seem to be about 30 minutes.

There is a possibility that some failures in the communication and tracking networks could be identified and repaired within a reasonable launch window. Here again, operating data relating frequency of occurrence and "off-line" times are necessary to determine a minimum useful launch window. These failures probably fall into two groups — those that are essentially momentary (less than 10 minutes), and those requiring at least 45 minutes to repair. The impact of the repair-or-replacement time above can be reduced by preparing two launch vehicles and two spacecraft on different pads at the same time. The effect of this procedure should be investigated.

A planned "hold" of perhaps 120 minutes could be employed to make greater use of the launch window (*Ranger* uses a planned hold of 75 minutes). The spacecraft has essentially no limitations in this regard since the gyros, which are started at T−60 minutes, have a 2000-hr life. The practical limits to this type of hold should be investigated.

Extensive project-wide system testing could also reduce launch window requirements.

2. Launch vehicle limitations. To date there is no *Atlas/Centaur* launch experience to draw upon. However, JPL has made seven launches using an *Atlas/Agena* launch vehicle. Countdown data for these launchings are presented in Appendix D. Six of the seven launches required 40-minute windows or less, and the seventh required a 60-minute window. It is anticipated that the *Atlas/Centaur* system will require launch windows at least as long as those required by the *Atlas/Agena.*

Effects. From Section II, reducing launch window duration will permit more launching days because days with less negative lunar declinations can be accepted. Also, the maximum launch window for the direct-ascent case is about 80 minutes, as compared to four hours for the parking-orbit case.

D. Transit Time

Current requirements specify a nominal transit time of 66-hr.

Minimum requirements are for either a 66-hr or a 90-hr-transit time; however, all data presented herein are for 66-hr-transit times. This requirement is based upon the following:

1. To date, all planning has been on the basis of a 66-hr-transit time.

2. 66-hr or 90-hr transit times are required to insure that the lunar landing occurs when observable from the Goldstone DSIF.
3. The spacecraft design and mechanization does provide capability for accomplishing missions with nominal 66 or 90-hr transit times.
4. The 90-hr trajectory requires a lower injection velocity—this could result in a 40- to 60-lb spacecraft weight reduction through saving in main retro fuel weight over and above the additional midcourse correction fuel that would be necessary. The launch vehicle fuel could also be reduced for the lower injection velocity.
5. The 90-hr transit time would be less reliable since the spacecraft must operate for a longer transit time.
6. Figures 3 and 6 are based upon a perigee outward radial central angle v_S of 170 deg, which corresponds to a 66-hr transit trajectory.
7. A 90-hr transit time has v_S = 175 deg, and increasing v_S has an effect like that of making the true anomaly more negative. That is, it permits a larger launch window for the same lunar declination. Appendix B presents the constraining relationships and launch calendar for 90-hr transit trajectories. Comparing Fig. 3 and Fig. B-1, it can be seen that for 66-hr trajectories a lunar declination of -21 deg is required for a 40-minute launch window, and for 90-hr trajectories 40-minute windows can be achieved at a lunar declination of only -18.5 deg. However, the landing site for normal descent is 45 deg W longitude for the 66-hr transit time and 65 deg W longitude for the 90-hr transit time. The minimum landing site requirements listed in III.F below limit the potential landing site to an area between 8 deg W longitude and 65 deg W longitude for the 66-hr transit time, and to between 28 deg W longitude and 65 deg W longitude for the 90-hr trajectory. This constraint has the effect of reducing the number of launch days available in some months for the 90-hr trajectory as compared to the 66-hr case due to post-landing lighting limitations. Comparing Figures 6 and B-2 indicates that the number of launch days which meet the minimum lighting requirements is about the same for both cases.

E. Landing Observability by DSIF

Current requirements specify that the lunar landing shall be observable by the Goldstone DSIF and that this visibility shall be maintained for a period of not less than three hours after landing.

Minimum requirements are the same as the current requirements above. The communication channel between Goldstone and the SFOF at JPL is much superior to that of the other two DSIF's in bandwidth and reliability.

Effects. In the worst direct-ascent case, which occurs in July 1965, the Goldstone post-landing visibility is 2.5 hours (see Appendix E).

F. Landing Location

Current requirements specify that the landing location for each mission shall be in a preselected region on the Moon's visible surface. The selenographic coordinates of the desired landing location shall be specified at least six months before launch. Typical nominal and limit landing regions are shown in Fig. 8.

Minimum requirements are that the spacecraft land in a maria-like area and that the angle between the unbraked approach-velocity vector and the local Moon vertical not exceed ± 25 deg. This requirement is based on the following:

1. The spacecraft has a design landing capability for approach angles up to ± 45 deg to the local Moon vertical; however, the probability of successful landing may be significantly reduced for approach angles greater than ± 25 deg.
2. The maria-type areas are believed to offer a higher probability for successful landing than other lunar areas.

Effects. The minimum requirements permit landing within ± 30 deg latitude for any longitude between 8 and 65 deg W (see Fig. 8). Selecting landing sites within this area to maximize post-landing sunlight adds significantly to the number of launch days available (see Appendix F). Figure 6 was prepared using this concept. Since the Moon rotates about 12 deg per day[4], increasing this constraint by 12 deg in longitude will add an additional launch day to some months. For the parking-orbit case, changing the "current requirement" to the "minimum requirement" would increase the number of available launch days from 8 to 16 for every month.

[4]Based on synodic period of 29.5 mean solar days.

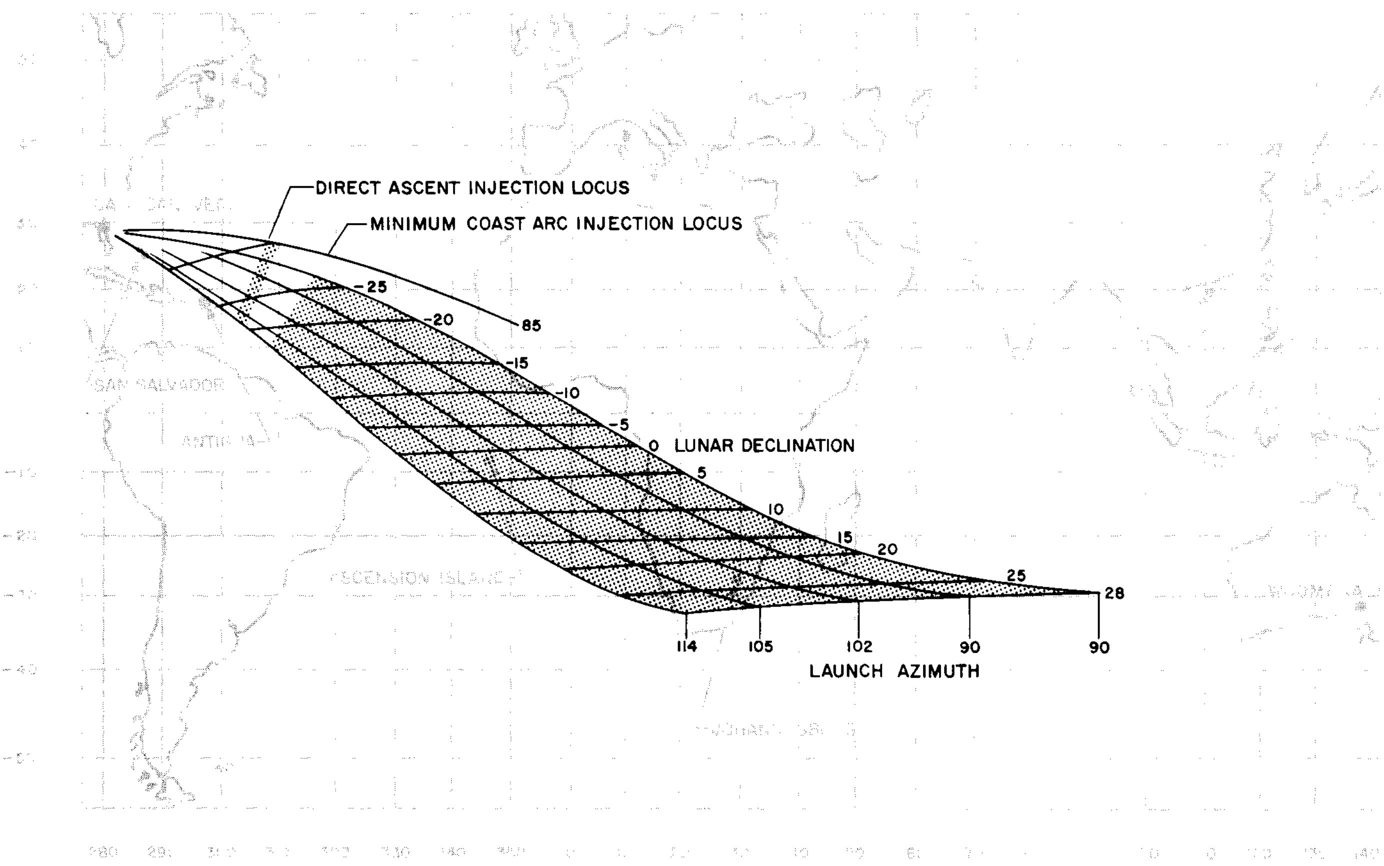

Fig. 8. Direct-ascent and parking-orbit injection locations

G. Lunar Lighting

Current requirements. The spacecraft shall approach the lunar surface in sunlight at least 24 hr after night/day terminator and have a minimum of 150 hr of post-landing sunlight before day/night terminator.

Minimum requirements. The spacecraft shall approach the lunar surface in sunlight at least ten hours after night/day terminator and have a minimum of 72 hr post-landing sunlight before day/night terminator. This requirement is based upon the following:

1. Approach TV pictures will yield valuable information relating to lunar surface characteristics, landing site location, and spacecraft functioning during the final descent phase (see Appendix G). To meet minimum photographic requirements, these pictures must be taken at least ten hours after sunrise (night/day terminator).

2. Approach TV pictures take on added significance if the spacecraft does not survive touchdown.

3. As presently designed, the spacecraft cannot survive more than about $\frac{2}{3}$ of a lunar night without additional battery power capability (see Appendix H).

4. If sufficient battery power were available, the post-landing TV has a probability of about 0.6 of surviving the lunar night and operating satisfactorily the next day.

5. Seventy-two hours are required to carry out a minimum acceptable scientific program (see Appendix I).

6. The landing site can be located to an accuracy of ± 2 to ± 8 km using approach TV, and of ± 3 to ± 150 m when the approach TV pictures are correlated with a post-landing TV survey; whereas, with tracking data alone, the location accuracy is expected to be ± 10 to ± 30 km.

Effects. The elimination of all lighting constraints would permit launching using direct-ascent trajectories on any day that the lunar declination is more negative than -20 deg, since -20 deg provides a minimum launch mum launch window of 30 minutes (see Fig. 3). This would result in about five launch days every month of the year.

Figure 6 shows the calendar of launch days when post-landing sunlight is not a constraint, and adding lighting constraints limits the number of launch days. It is noted that March and April in 1965 and 1966 are the only months where there is a significant increase in launch days available when the lighting is reduced from 150 hours to 72 hours.

H. Launch Opportunities Per Launch Period

Current requirements specify that there shall be at least eight launch days in each launch period.

Minimum requirements. Each launch period should have at least five launch days with acceptable launch windows. This requirement is based upon the following reasoning: The launch-vehicle/spacecraft system can be readied for launch on alternate days. Five launch days would then permit about three countdowns in a launch month. This should result in a minimum acceptable probability for successfully getting the launch vehicle and spacecraft system off the pad in a launch period—perhaps 0.5 to 0.7. Table 1 summarizes from Appendix D the *Ranger* and *Mariner* experience.

Table 1. JPL launch experience

Spacecraft	Countdown days used	Launch window used, minutes
RA-1	5 + 1[a]	31
RA-2	5 + 1[a]	35
RA-3	1	5
RA-4	1	39
MR-1	2	36
MR-2	3	65
RA-5	3	21

[a]Indicates 5 unsuccessful countdowns in the first launch period, and one countdown in the second launch period which resulted in launch.

Effects. Parking-orbit trajectories provide more launch days per launch period than do direct-ascent trajectories. For the minimum requirements described under each constraint above, the former has 16 launch days every month, whereas the latter has a maximum of five days in some months. Reducing the five-day requirement would increase the number of launch periods available.

I. Number of Missions Desired Per Year

Current requirements are for two launchings in 1964 and four or five per year thereafter.

Minimum requirements appear to be the central decision that must be made before the feasibility of direct-ascent trajectories can be determined. Table 2, summarized from Section II, lists the months which meet the minimum requirements listed under the constraints above (III. A - H), for the period between July 1964 and September 1966.

Table 2. Launch periods, 1964-1966

1964		1965		1966	
Month	Launch days available	Month	Launch days available	Month	Launch days available
Jul	4	Jan	0	Jan	0
Aug	5	Feb	2	Feb	2
Sep	3	Mar	5	Mar	4
Oct	1	Apr	6	Apr	6
Nov	0	May	5	May	6
Dec	0	Jun	5	Jun	5
		Jul	6	Jul	6
		Aug	5	Jul/Aug	6
		Sep	4	Aug/Sep	5
		Oct	3	Sep	4
		Nov/Dec	0		

The number of possible launches per year is determined by the necessary spacing between launchings (2-month or 3-month centers) as well as the other constraints previously discussed. Table 3 indicates the number of launches available for parking-orbit and direct-ascent trajectories for various constraints. "Launch days per period" indicates the number of successive days that are necessary to assure some probability of making a successful launching. The probability of successful launching will increase as the number of launch days per period is increased. It is recognized that the winter months have wind conditions that may preclude launching on some days, but since 8–16 days are available for the parking-orbit case, it has been assumed that wind conditions will not prevent launching during any particular chosen month.

Effects. Direct-ascent trajectories have a reasonable probability of permitting two or three launchings per year without compromising the achievement of the basic project objectives. More than this number of launchings is not feasible using direct-ascent trajectories. Parking-orbit trajectories will permit four to six launchings per year and each with a higher probability than the direct-ascent case.

Table 3. Launch opportunities per year

Trajectory Mode	6 to 8 launch days per period[a]		6 to 16 launch days per period[b]		5 launch days per period[b]		4 launch days per period[b]	
	2-mo. centers	3-mo. centers	2-mo. centers	3-mo. centers	2-mo. centers	3-mo. centers	2-mo. centers	3-mo. centers
Direct-ascent	0	0	0	0	3	2	4	3
Parking-orbit	6	4	6	4	6	4	6	4

[a]Current constraints as discussed in this Section.

[b]The following minimum constraints:

1. Minimum 72-hr post-landing sunlight.
2. Landing area varied between 8°W and 65°W.long. to maximize post-landing sunlight.
3. 66-hr transit trajectories.
4. True anomalies −5° to +13°.
5. Launch azimuth 83° to 114° E of true North.
6. Minimum 40-minute launch window required to qualify a launch day.

IV. SUMMARY

A comparative summary of advantages for the project, the spacecraft, and the launch vehicle, for direct-ascent and parking-orbit trajectories, based on the following minimum requirements (except as noted) concludes this study.

The minimum requirements established for the summary are the following:

1. Lunar approach in sunlight, post-landing sunlight 72 hr minimum.
2. Launch window 40 minutes minimum.
3. Landing location varied from 8 deg W to 65 deg W to maximize post-landing sunlight.
4. Launch azimuth: 83 – 114 deg E of true North.
5. For direct ascent, true anomaly: -5 to $+13$ deg.
6. Transit time: 66 hr.
7. Launch days per period: minimum of 5.

The adoption of the minimum requirements described herein probably would not require spacecraft redesign; however, it would require a maximum of pre-launch system testing since there would be fewer launch days available.

Computation techniques and trajectories would have to be developed for direct-ascent trajectories.

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Advantages of Direct Ascent	**Advantages of Parking Orbit**
PROJECT	
Launch schedule: possible gross improvement through shortening of launch vehicle development time (q.v.).	Launch schedule: generally more flexible, in number of missions per year (4-6 vs 2-3), number of days per month (8-16 vs 0-5), length of window (50-240 minutes vs 80 minutes max).
Launch vehicle payload, including spacecraft: possible 50-75 lb increase (see III.A).	Landing site: may be preselected for an 8-day launch period (see Fig. 3).
Mission mode: inherently simple, provides for advance in direct-ascent art, but requires training of operations personnel.	Mission mode: permits continuity with *Ranger*, *Mariner*, previews *Surveyor* planning, trajectories, requires no retraining.
LAUNCH VEHICLE	
Development and engineering simplification: no *Centaur* sun seeker, tracking and telemetry simplification, possible reduction in other launch vehicle requirements (see Appendix J).	Development and engineering: provides continuity with previous *Centaur* development, requires *Centaur* sun seeker (for anti-boiloff maneuver) only for coast > 20 minutes, results in more versatile vehicle usable for interplanetary, synchronous-satellite, other missions.
Development time: possible decrease.	
Operations: no restart.	Operations: less chance of aerodynamic heating.
SPACECRAFT	
Weight: possible increase (see III.A).	
Tracking: possible reduction in AMR tracking and telemetry requirements, since injection occurs over a narrow band (see Fig. 8).	
	Mission mode: no spacecraft redesign required, provides continuity with *Ranger*, *Mariner*, previous *Surveyor* development.
	Landing: normal approach (at 45 deg W long., 8-day period) may always be selected for increased reliability.
	Operations: possibility of spacecraft aerodynamic heating and other effects minimized.

APPENDIX A

Glossary

The terms used in this document are rather specialized and require a working knowledge of lunar trajectories for a full comprehension. The purpose of this Appendix is to give brief descriptions of some of the terms and concepts used.

Ascent Trajectory—That phase of flight from launch until the time when the spacecraft is essentially travelling radially outward from Earth—in lunar missions, approximately the first two hours after launch.

Burning Arc—The arc subtended at the Earth's center in the trajectory plane during launch vehicle burning (Fig. A-1, A-2).

Direct-Ascent Trajectory—The method of achieving the required injection energy by burning the stages successfully and continuously, allowing only sufficient coasts (or small ullage-propulsion phases) between stages to successfully accomplish the staging separation sequences (Fig. A-1, A-2).

Launch Period—The number of *days* during which launch can occur.

Launch Window—The number of *minutes* during a launch period permitted for launching.

Lunar Day—The lunar day is equal to about 14½ Earth days.

Lunar Declination—The angle between the Earth-Moon line and the projection of this line on the Earth's equatorial plane. The variation in lunar declination during any given month depends on the year. Thus, in 1960, declination of the Moon varies between ±18.5 degrees, in 1965 it varies from ±26 degrees, and in 1969 it varies between ±28.4 degrees. The maximum

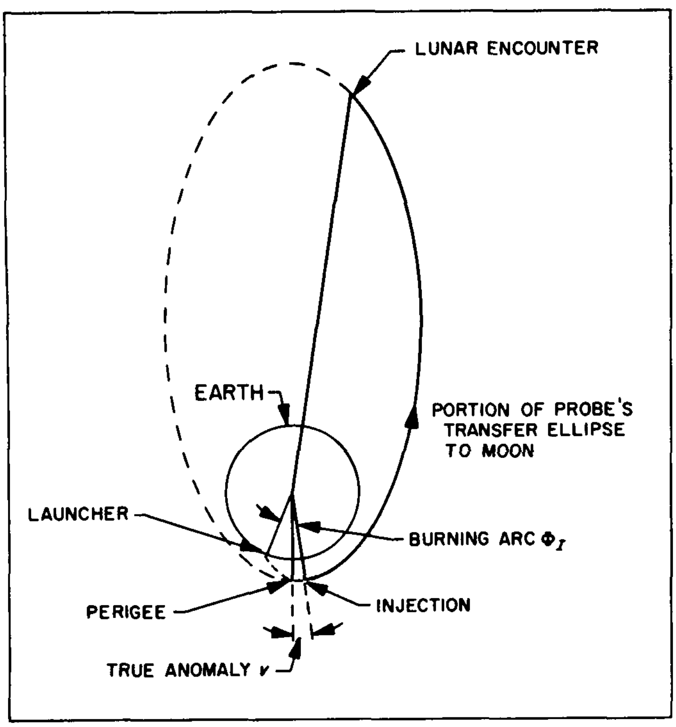

Fig. A-1. Direct-ascent lunar trajectory geometry

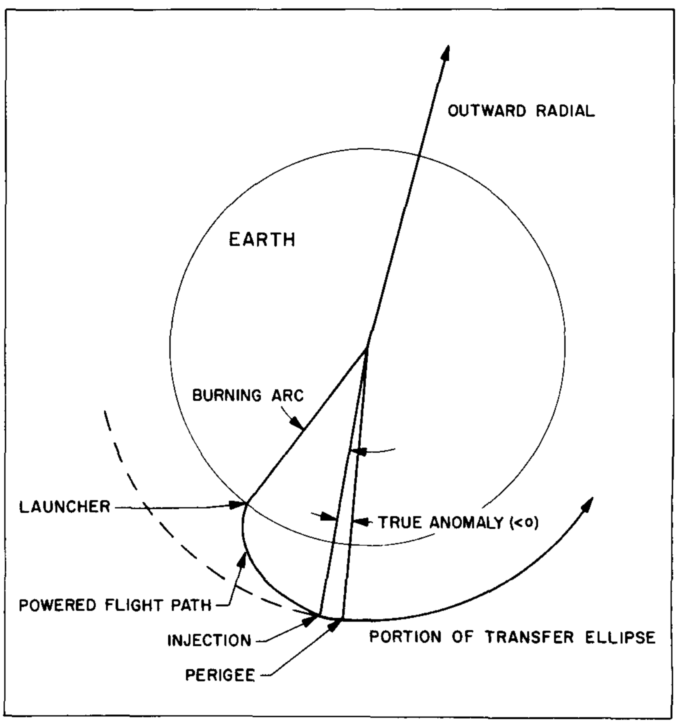

Fig. A-2. Ascent trajectory, direct ascent absolute variation in lunar declination is from 18.5 to 28.4 degrees and has a 9½ year cycle.

Parking-Orbit Trajectory—The method of achieving the required injection energy by dividing the burning phase into two powered arcs separated by a circular coasting interval or parking orbit (Fig. A-3).

Terminator—The line on the surface of a celestial body between light and dark, i.e., the shadow line.

True Anomaly—The angle subtended at Earth's center between perigee and injection of the lunar transfer ellipse (Fig. A-1, A-2).

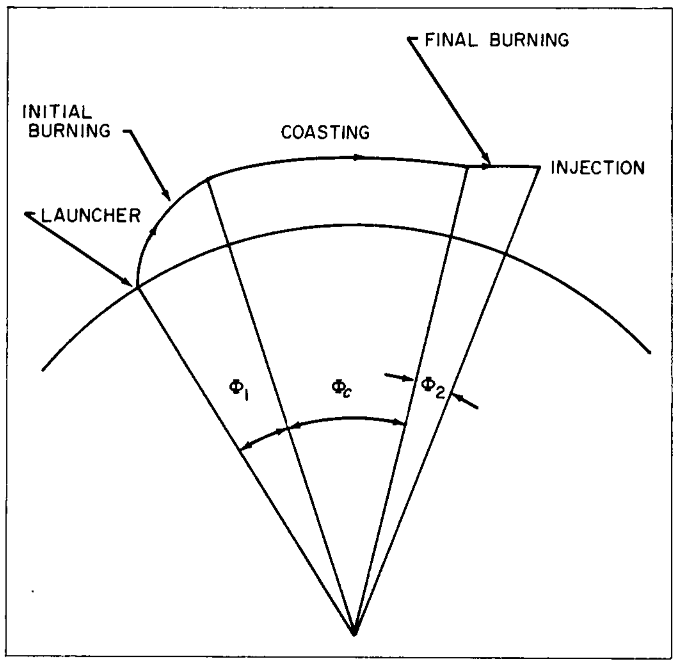

Fig. A-3. Parking-orbit burn arc

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APPENDIX B

Launch Calendar for 90-hour Transit

The two Figures in this Appendix are analogous to the two major Figures in Section II, with the change in transit trajectory from the 66-hr to the 90-hr transit. Figure B-1 corresponds to Fig. 4, and Fig. B-2 corresponds to Fig. 6, with the following exceptions:

1. Launch days which provide less than 72-hr post-landing sunlight are not shown.

2. Launch windows show an arbitrary ceiling of 50 minutes, corresponding to a true anomaly between −5 and +6 deg. If true anomaly were increased to +13 deg, the maximum window would correspond to that shown in Fig. 3 (and the peaks shown in Fig. 6).

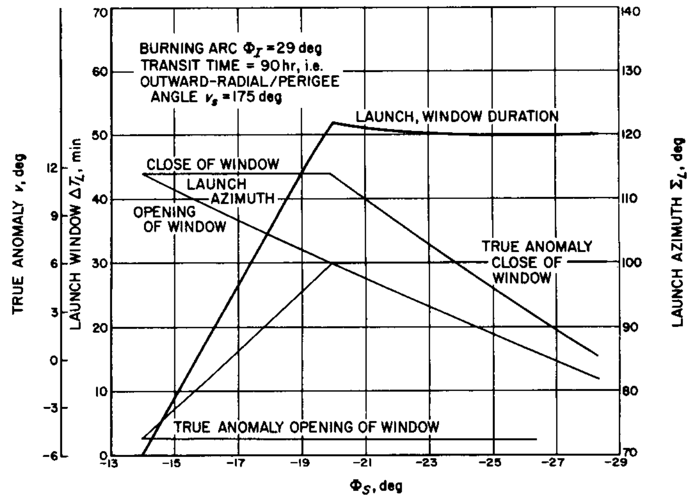

Fig. B-1. Launch window, launch azimuth, true anomaly vs lunar declination for 90-hr transit

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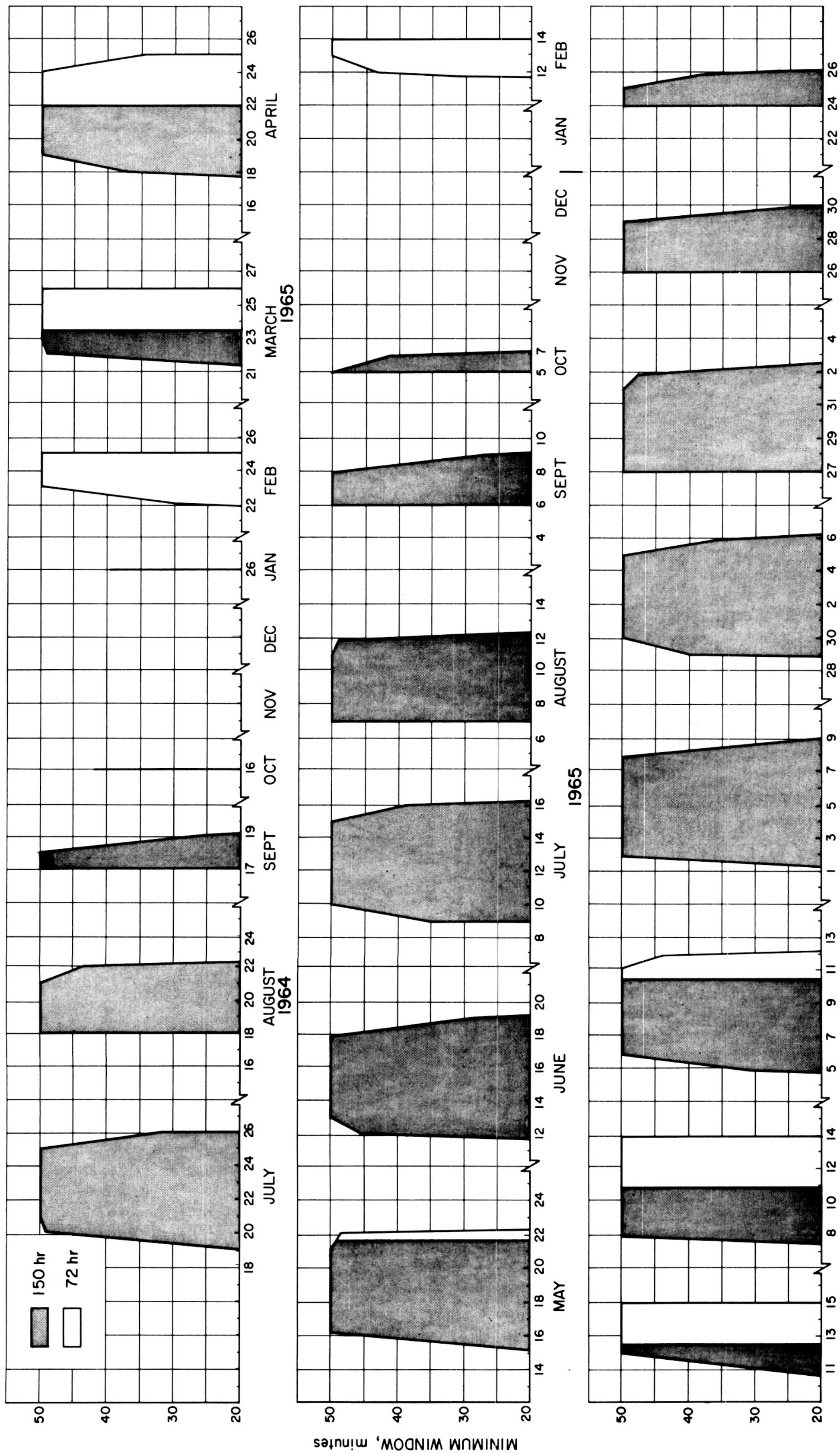

Fig. B-2. Launch calendar for 90-hr transit

APPENDIX C

Implications of Increased Midcourse Fuel

I. BACKGROUND

With a given permissible range for main-retro burnout velocity (presently determined by radar sensor considerations), an increase in midcourse propellant weight allotment reduces launch capability when both lunar perigee and lunar apogee conditions are considered unless the additional weight is compensated for by ballasting.

The present status of the *Surveyor* burnout velocity is as follows:

1. The highest value of *nominal* burnout velocity that can be tolerated is 575 ft/sec. Under this condition, the highest burnout velocity may reach 700 ft/sec in the 3 σ sense. In the design, we always choose this nominal point to correspond to the simultaneous conditions of (a) highest unbraked impact velocity (V_{impact}) and (b) zero midcourse correction. At present, *linear* doppler performance is limited to 700 ft/sec at the upper end.
2. The lowest value of *nominal* burnout velocity is fixed at 200 ft/sec. This roughly satisfies a requirement of no greater than 1 percent probability of exceeding doppler sensor performance at the low velocity end. The low velocity limit arises because of (a) larger flight path angles as nominal velocity decreases and lateral velocity dispersions remain constant and (b) increasing probability of zero or negative doppler in one or more beams for the same reason as (a). The 200 ft/sec condition is matched to the combination of the lowest unbraked impact velocity and 3 σ midcourse correction.
3. The difference in burnout velocity due solely to weight variation at ignition (corresponding to the conditions of zero and 3 σ midcourse correction) amounts to 132 ft/sec.
4. The net allowable variation in unbraked impact velocity is therefore $575 - 200 - 132 = 243$ ft/sec $= 74$ m/sec.
5. In studying impact velocity variations for different lunar geometries, it has been found that the required spread in unbraked velocity (if launch capability is not to be affected) is from 2595 to 2686 m/sec. A total spread of 91 m/sec is therefore necessary.
6. Currently, the deficiency, 17 m/sec (91-74), is to be made up by suitable ballasting at the low impact velocity condition (lunar perigee). This increases the spacecraft weight to retro propellant ratio, and, in turn, increases the lunar impact velocity. For an eight-day launch period within any given month, the impact velocity variation is generally much smaller than 91 m/sec. Thus, if the velocity tends to be on the low side within that month, a fixed amount of ballasting can be used to intentionally increase the burnout velocity across the board. The maximum amount needed is 9.2 lb and will be in the form of additional vernier propellant. It should be emphasized that this additional propellant is *not* available for use during the midcourse correction, for that would defeat the purpose of ballasting.

II. ADDITIONAL MIDCOURSE CORRECTION CAPABILITY

Since the present vernier propellant requirement (153.9 lb total loading) leaves approximately 20 lb of unfilled tank capacity, some additional midcourse correction can be gained by permitting the tanks to be filled.

If the unbraked impact velocity limits remain 91 m/sec or 298 ft/sec apart, the division of any incremental vernier propellant between midcourse and ballasting may be calculated.

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The velocity spread required is made equal to the velocity spread available, or:

$$V_{22} + x\frac{\partial V_{BO}}{\partial W} + V_I = y\frac{\partial V_{BO}}{\partial W} + S_I$$

where

- x is the additional midcourse (vernier) propellant allotment, lb
- y is the additional ballast allotment in the form of vernier propellant (to be used only at the low-impact-velocity end), lb
- $\frac{\partial V_{BO}}{\partial W}$ is the sensitivity of burnout velocity with respect to initial weight = 6.0 ft/sec lb
- V_I is the variation in impact velocity between what perigee and apogee = 298 ft/sec
- S_I is the allowable spread in unbraked velocity = 375 ft/sec

and $x + y$ = total additional vernier-propellant loading = 20 lb

Substituting and solving

$$y = 14.6 \text{ lb}, \qquad x = 5.4 \text{ lb}$$

In this case, if vernier fuel is used for ballasting at the low impact velocity end (lunar perigee), only 5.4 lb would be available for additional midcourse correction.

With the additional 5.4 lb, the total of 27.4 lb of midcourse fuel would allow a 38 meters/sec velocity correction. This provides an FOM of 12.7 meters/sec (1 sigma).

A further consequence of increasing the midcourse allotment is that the main engine propellant must be increased somewhat to maintain a nominal burnout velocity of 575 ft/sec under the simultaneous conditions of zero midcourse correction and highest impact velocity. Corresponding to the above conditions, we find that 7.7 lb of additional solid propellant is needed in order to offset the equivalent dead weight increase of 5.4 lb. Thus, the total separated weight is higher by 13.1 lb at the high velocity and 27.7 lb at the low velocity end.

If vernier fuel is not used for ballast, and external ballast is used, more vernier fuel can be used for midcourse correction.

$$y = x + 9.2 \text{ lb}$$
$$\text{If } x = 20 \text{ lb, then } y = 29.2 \text{ lb}$$

(1.43 lb of retro propellant is required for each additional pound of spacecraft dead weight.)

Under these conditions the total separated weight is increased by 20 + 20 (1.43) = 48.6 lb at the high velocity, and increased by 48.6 + 29.2 = 77.8 lb at the low velocity end. It is recognized that the launch vehicle can inject perhaps 40 pounds more weight at the lower velocities so the 48.6 pounds is the critical weight addition. With 20 lb additional vernier fuel the total of 42 pounds of midcourse fuel would allow 42 × 1.39 = 58 m/sec (3 sigma) = 19.3 m/sec (1 sigma). This assumes that the ballast weight can be attached to and be jettisoned with the main retro case. If this cannot be done, additional vernier fuel would be required during the final descent phase (perhaps 10% of the additional fuel).

APPENDIX D

JPL Launch-on-Time Experience

As a source basis for considering *Surveyor* prelaunch and launch operations durations vis-a-vis the launch window, a record of JPL experience with *Ranger 1*-5 and *Mariner 1*-2 launches is presented in Fig. D-1, with explanations keyed to events shown on the Figure presented in the outline below.

Ranger 1

1. The first attempt, on 7-29-61, was scrubbed at T−27 minutes because of failure of the industrial power on the Cape.
2. The second attempt, for a 7-31-61 launch, was scrubbed prior to entering the Atlas portion of the countdown because of a leak in the spacecraft attitude control system.
3. The third attempt, on 8-1-61, was scrubbed at T−15 minutes when lox transfer valve LB1 stuck open after detanking lox to investigate an Agena helium tank low pressure indication.
4. The fourth attempt, for an 8-2-61 launch, was scrubbed prior to entering the Atlas portion of the countdown because the spacecraft pyrotechnics were inadvertently fired.
5. Twenty-eight minutes at T−50 minutes to allow service tower removal, and to perform checks on the Atlas displacement gyro torquing circuitry.
6. Nine minutes at T−5 minutes because of a guidance temperature problem.

Ranger 2

1. 20 October, the test was aborted after the autopilot guidance loop test. The "initiate separation sequence" discrete did not reach the autopilot programmer and the problem could not be resolved. The problem was later found to be due to an open circuit in a three-way splice; wire ZB131A22 to pin A of plug 301U3P1 on the GE guidance decoder had pulled out of the splice.
2. 23 October, the test was terminated because of hydraulic fluid leakage at the V2 engine.
3. 24 October, the test was scrubbed at the request of LMSC, Sunnyvale, due to hydraulic problems observed on other Agena vehicles and an Agena captive firing.
4. At T−60 minutes for 88 minutes. The first 70 minutes were planned to meet the launch window requirements. The 70-minute planned hold was extended 18 minutes due to service tower replacement around the missile for an Agena guidance check.
5. At T−31 minutes for 4 minutes because of a frozen valve in the lox tanking system.
6. At T−5 minutes for 18 minutes because of oscillations on the Gilmore weight digital readout panel during subcooled lox topping. The actual lox level was determined by slightly raising the lox tank pressure to dampen the oscillations.

Ranger 3

1. At T−205 minutes for 30 minutes to complete Atlas engines igniter installation.
2. At T−60 minutes for 40 minutes to meet launch window requirements. A 70-minute hold had been planned, but 30 minutes were consumed by the hold at T−205 minutes.
3. At T−5 minutes for 10 minutes. The first 5 minutes were planned to meet launch window requirements. The planned 5-minute hold was extended 5 minutes to complete lox topping.

Ranger 4

1. At T−60 minutes for 85 minutes. The planned 70-minute hold was extended for 15 minutes as a result of umbilical P1005 being inadvertently knocked out early in the hold. The pneumatic panel differential pressure warning light came on, sensing a false zero bulkhead differential pressure, and subjected the missile to Sequence II pressures. The umbilical was reinstalled and restep to Sequence I pressures was satisfactory.
2. At T−40 minutes for 6 minutes to complete GE evaluation of loop test No. 2.
3. At T−15 minutes for 8 minutes to complete lox tanking.

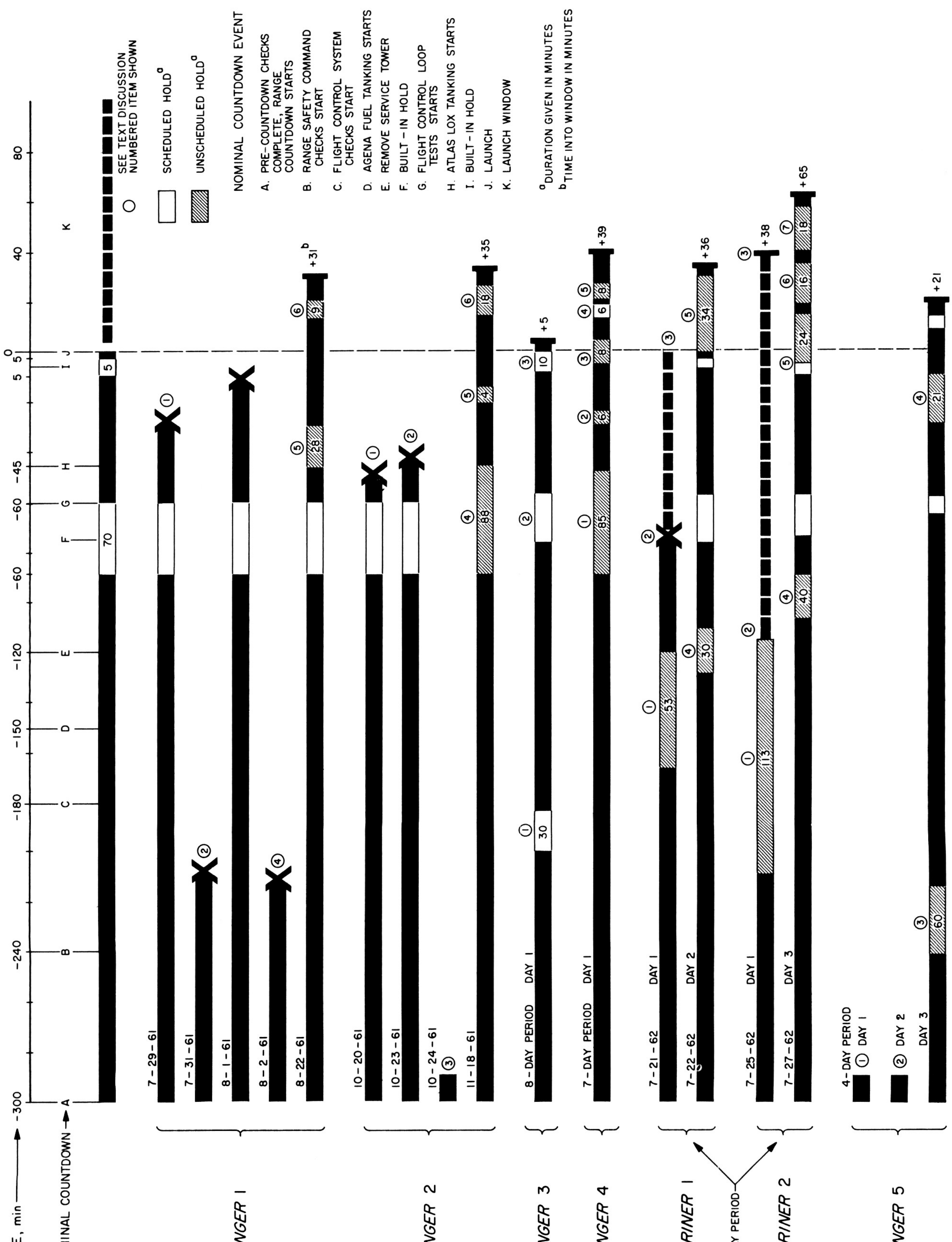

Fig. D-1. JPL launch experience

4. At T−5 minutes for 6 minutes as planned to meet launch window requirements. Launch Plan 23G was established.
5. At T−2 minutes and 27 seconds for 8 minutes due to a GE guidance redline callout. The count was immediately recycled to T−5 minutes. The ground guidance track transmitter power was low due to a faulty cabinet door interlock in the guidance ground station. This situation was corrected and GE guidance reported a 'go' condition at 1540 EST. The launch plan was revised to 23H and the countdown was resumed at 1545 EST, proceeding through launch without further difficulty.

Mariner 1

1. 53-minute hold at T−165 to complete Range Safety Command checks.
2. Flight scrubbed at T−79 due to inability to solve RSC problem.
3. If count had continued from T−79 in nominal times, lift-off could have been at beginning of window.
4. 30-minute hold at T−130 to change Azusa transponder.
5. 34-minute hold at T−80 sec for GE Guidance. Count recycled to T−5 minutes at end of hold and launched.

Mariner 2

1. 113-minute hold at T −205 to complete Range Safety Command checks.
2. Flight scrubbed at T−205 due to inability to solve RSC problem.
3. If count had continued from T−205 in nominal times, lift-off could have been after a 38-minute penetration into the launch window.
4. 40-minute hold called at T −100 to replace *Atlas* battery.
5. 24-minute extension of G-5 scheduled hold due to GE Guidance problem.
6. 16-minute hold called at T−60 sec due to GE Guidance problem. Count recycled to T−5 and resumed.
7. 18-minute hold called at T−50 sec due to GE Guidance problem. Count recycled to T−5 minutes and resumed.

Ranger 5

1. Due to spacecraft problems encountered prior to the first launch day, the attempt was postponed for one day.
2. Launch attempt rescheduled for the third day of this period because of an unfavorable weather prediction (Hurricane Ella).
3. 60-minute hold at T−245 to replace power supply and voltage regulator in *Agena* telemetry system.
4. 21-minute hold at T − 25 for further evaluation of wind conditions and completion of lox tanking in *Atlas.*

APPENDIX E

Goldstone Visibility Tradeoffs

Goldstone post-landing visibility may be increased at the rate of 1.83 minutes for each ft/sec increase in unbraked-velocity-handling capability.

The velocity-handling capability can be affected in three ways:

1. Keep spacecraft injected weight constant, decrease spacecraft payload weight and increase main retromotor weight. This results in a velocity-handling capability of 10.75 ft/sec per lb increase in main retro (and corresponding weight decrease in spacecraft payload).

   $\frac{60}{1.83\ (10.75)} = 3.05$ lb/hr increase of Goldstone visibility.

2. Reduce injected weight and maintain main retromotor weight constant. This results in a velocity-handling capability of 6.46 ft/sec per lb decrease in injected weight.

   $\frac{60}{1.83\ (6.46)} = 5.07$ lb/hr increase of Goldstone visibility.

3. Maintain constant spacecraft payload, increase main retromotor weight. This results in a velocity-handling capability of 4.45 ft/sec per lb increase in main retromotor weight.

   $\frac{60}{1.83\ (4.45)} = 7.4$ lb/hr increase of Goldstone visibility.

APPENDIX F

Post-Landing Sunlight

Figure F-1 shows the relative location of the Sun, Earth and Moon for 22–27 March 1965.* For 22 March the angle between these three bodies (LPH) is 242 deg, and increases approximately 12 deg for each succeeding day as shown. The LPH angle for sunrise and sunset for a landing location at 45 deg W longitude is also indicated. Since lunar impact on 22 March occurs after sunrise at 45 deg W longitude, the landing will be in daylight and the time to sunset is determined by the difference between the arrival LPH (242 deg on 22 March) and the sunset LPH (315 deg for 45 deg W longitude). The Moon's angular rate is about 360 deg ÷ 29.5 (24) = 0.5 deg/hr. Time from landing to sunset for this case is

$$(315 - 242) \div 0.5 = 146 \text{ hours}$$

Figure F-2 shows the relationship between post-landing sunlight available vs lunar landing longitude for 22–27 March 1965 arrival dates. These dates are those wherein the lunar declination is sufficiently negative to

*All dates given here are *arrival* dates.

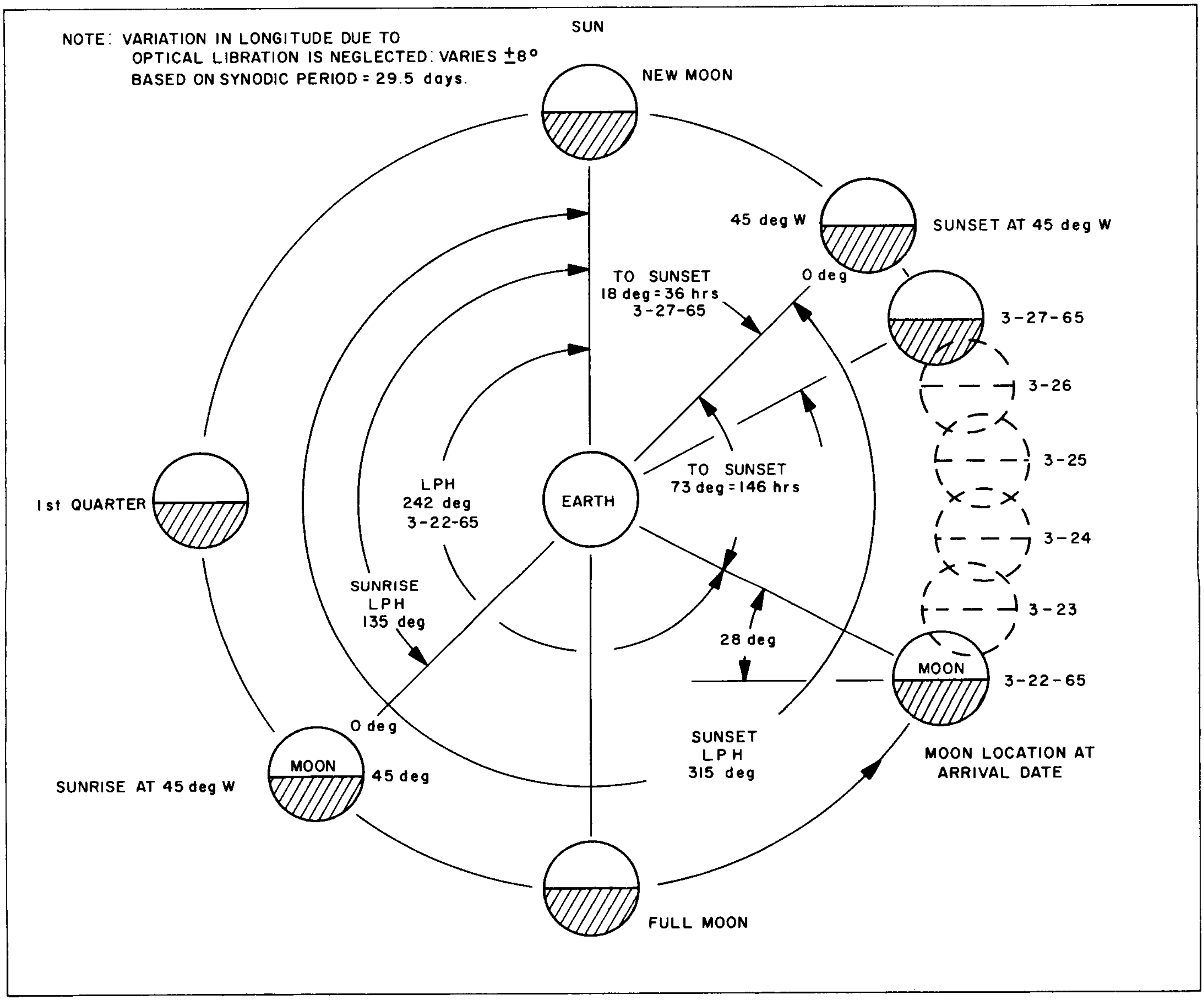

Fig. F-1. Lighting geometry, 22-27 March 1965

allow direct-ascent trajectories. (See basic report for constraints). For this case maximum post-landing sunlight is available at 65 deg W longitude and varies between 185 hours and 72 hours over the launch period.

Figure F-3 indicates the relative location of the Sun, Earth and Moon for 6–12 August 1965. In this case, all landings at longitudes greater than 22 deg W will be in the dark on 6 August. If the 60 deg W longitude landing is desired, landings could only be made on 10, 11 and 12 August.

Figure F-4 shows the relationship between post-landing sunlight available vs lunar landing longitude for 6–12 August 1965. An important difference between this Figure and Fig. F-2 is the "land in dark"/"land in light" line. In this case, the maximum post-landing lighting (about 360 hours) can only be realized by changing the landing longitude each day. On the other hand, if 15 deg W longitude is selected, the post-landing sunlight available would vary between 345 hours and 200 hours over the launch period.

When all arrivals are in sunlight, as in Fig. F-2, for a particular touchdown longitude the number of possible launch days can be increased by decreasing the post-landing sunlight requirements. When some arrivals are in darkness as in Fig. F-4, the number of possible launch

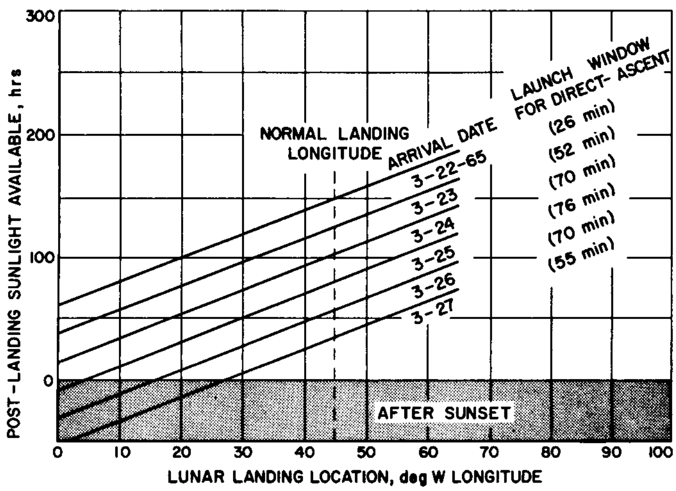

Fig. F-2. Post-landing sunlight, 22-27 March 1965

days can also be increased by moving the touchdown longitude.

An analysis of Fig. F-2 indicates that landing at 65 deg W longitude in March 1965 is the only case that will meet the stated 72-hr minimum post-landing lighting constraint (See basic report). On the other hand, Fig. F-4 indicates that landings at longitudes between 8 and 20 deg W are the only cases that will satisfy the lighting constraints for all launch days. If all landings are restricted to the normal approach case, only three days in March and four days in August would satisfy the lighting and launch window (minimum 40 minutes) constraints. Thus, the number of launch days in a particular month can be increased by "being able to vary the landing location between 8 and 65 deg W longitude to maximize the post-landing sunlight."

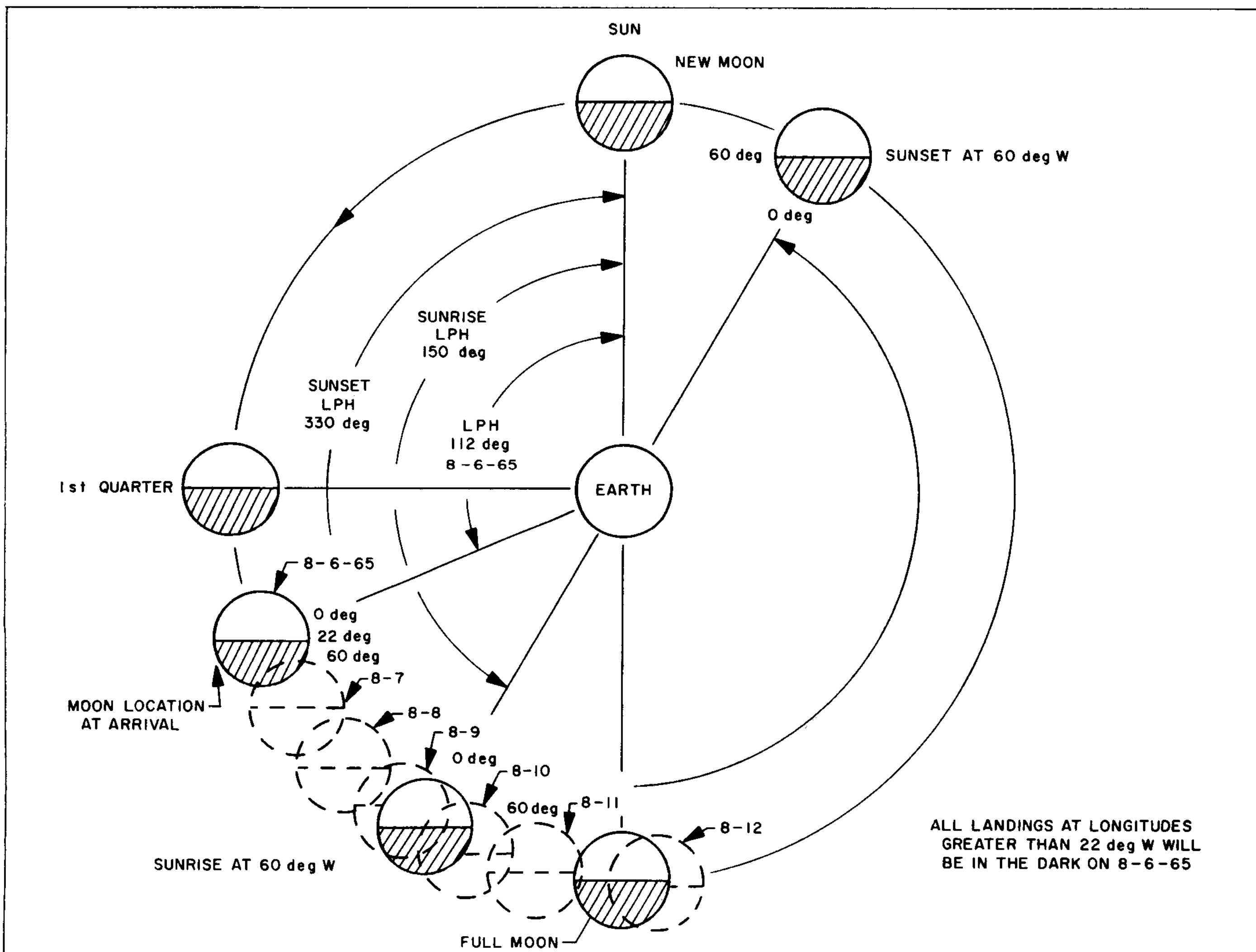

Fig. F-3. Lighting geometry, 6-12 August 1965

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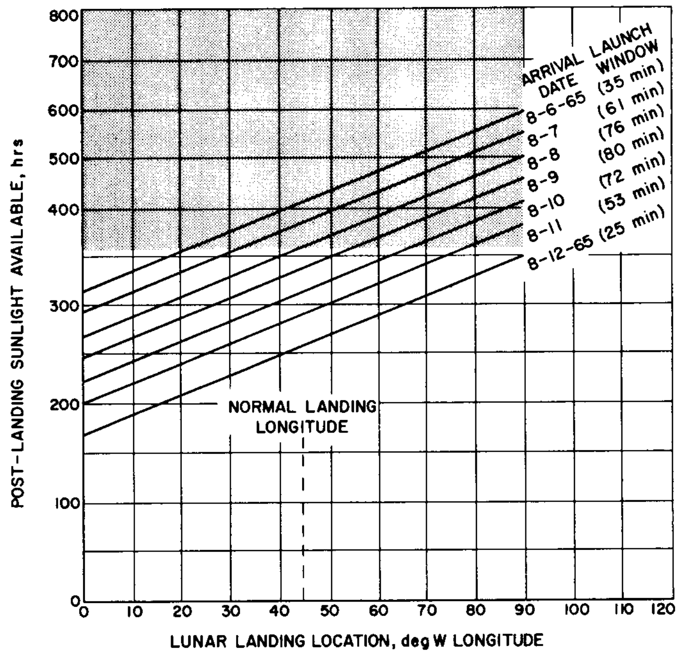

Fig. F-4. Post-landing sunlight, 6-12 August 1965

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APPENDIX G

Surveyor Approach Television

Television coverage of the lunar surface during the descent phase is provided in *Surveyor* as a means of precise determination of the landing site and to obtain photographs of the lunar surface in the landing area at a higher resolution than is obtainable from Earth observations. Various factors limit the extent to which these objectives can be achieved. They arise mainly from the fact that the primary *Surveyor* mission is to achieve a soft landing. Thus, the spacecraft configuration, attitude control, engine design, etc., are not optimized for descent television coverage. In addition, important engineering data, both to document the descent performance and for initiation of emergency procedures, preclude unlimited use of the TV which occupies the full transmission bandwidth of the telemetry system.

The various phases of the Surveyor descent from the point of view of television coverage are illustrated in Fig. G-1. The television sequence (Phase 1) begins at an altitude of about 1000 miles. This initial altitude is limited by the drift rate of the inertial attitude reference system which is used from the beginning of the terminal attitude maneuver until the doppler radar takes over in Phase 5. During Phase 1 a nominal 100 pictures will be taken to provide good correlation between the first and last frames. The lower altitude limit of this phase is set at about 80 miles by the uncertainty in the altitude as determined from tracking data relative to the actual altitude at which the marking radar signal occurs. A margin of about 20 miles is required to ensure that the TV will be turned off in time to receive verification of the altitude mark for possible initiation of emergency procedures. Thus, during Phase 2, telemetry from the spacecraft will be confined to engineering data.

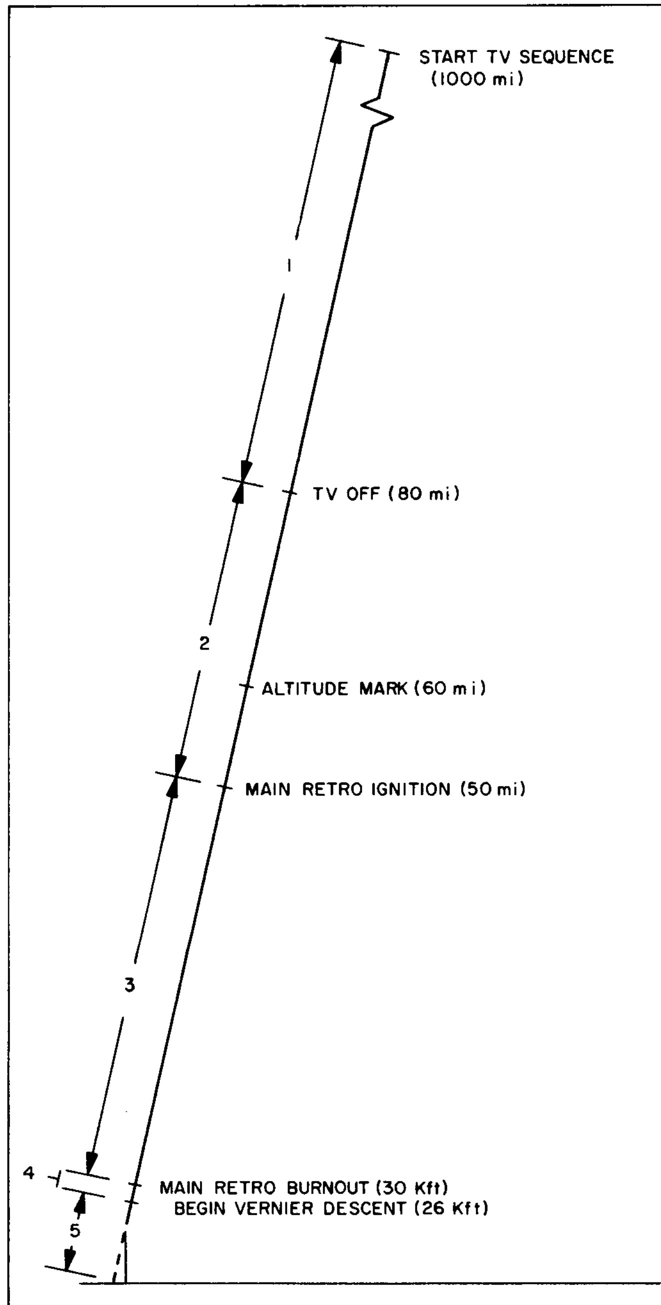

Fig. G-1. Descent-phase events

During Phase 3, main-retro burn, two pictures are presently scheduled. Although no definitive tests of the ability to view through the main-retro exhaust have been made, it appears very unlikely that pictures taken in this interval will be of significant quality. Phase 3 telemetry will be limited to essentially engineering data.

Phase 4 consists of an eight-second interval between main-retro burnout and separation. Since the spacecraft attitude remains constant during this interval, pictures can be taken subject to interference from the vernier engine exhaust. However, the command sequence required to permit telemetry of the burnout signal and separation data is such that only one frame can be taken. Even so, there is a risk of losing engineering data which will indicate the presence of disturbances introduced into the flight control system and could aid in determining the quality of separation. This is because there will not be time to verify that each command is received by the spacecraft before the next is sent or to send more than one of each command. An additional potential problem is that of contamination of the camera lens by the main-retro exhaust. Again, vacuum firings have not yet determined the extent of exhaust deposits, but it is presently assumed not to be significant. This problem could be overcome by providing a transparent jettisonable dust cap at some weight penalty.

Phase 5, vernier descent, does not appear favorable for TV transmission because of high-gain-antenna pointing errors. Even in a vertical descent it is expected that the lateral velocity resulting from main-retro thrust misalignment will cause attitude corrections of sufficient magnitude to greatly reduce the transmission bandwidth. However, several frames will be programmed for this phase of the descent.

The characteristics expected for pictures taken in Phases 1 and 3 are shown below:

Slant range, mi	Optical resolution, ft	Recognition capability, ft
1000	3000	9,000 – 60,000
80	200	600 – 4,000
7.6	14	42 – 280

The recognition capability for a particular optical resolution depends upon the nature of the target, the use of the data, the number of pictures available, and the data processing technique employed. In general, recognition capability is about 4 to 20 times the optical resolution; perhaps with a redundancy of pictures and advanced data processing, this number will approach 1.

The approach TV requires that the spacecraft land at least 48 hr before the day/night terminator and at least 24 hr after night/day terminator to provide satisfactory pictures (constraints F4 stop and 20 ms shutter speed).

APPENDIX H

Some Aspects of Night Television

A. Moon Shadow Constraint

The spacecraft design is such that it cannot be in the Moon's shadow for more than 30 minutes before landing. The gyro drift specification is 1 deg/hr and the design budget to keep the lateral velocity to within 150 ft/sec on a 3-sigma basis is ½ deg. Thus, the maximum time allowable to be on gyro control is 30 minutes. The effect of this 30-minute requirement for night landing condition is to reduce the number of calendar days available for landing by about three days per month if one desires to select a specific landing site. When the longitude variations in landing are ±20 deg or more and the launching days are chosen to maximize possible landing sunlight, this 30-minute restriction has no effect on the number of launch days available.

The thermal time constant for a number of the subsystems such as the landing shock absorbers, the vernier engines, the guidance and control, and the space frame also have about a 30-minute limitation. (It is interesting to note that 30 minutes in the shade will cool the space frame by approximately 30°F and that this cooling will reduce the spacecraft structural design margin from about 25% to 20%).

B. TV After Landing At Night

It may be possible to take TV pictures during portions of the lunar night with the present *Surveyor* TV system utilizing Earthlight (about 1.4 ft-candles). Under these conditions the picture obtained would probably be better than that obtained when using the daylight TV emergency mode. Operating the TV at night would require about 80 watts for heating and about 70 watts for operating. This total of 150 watts includes the transmitter power requirements. Without night TV, 9 watts are required to keep the TV system warm. Some mechanical redesign would be necessary to enable the mirror to operate during the lunar night.

C. *Power Requirements and Lunar Night Survivability*

As currently planned the spacecraft should land with about 2800 watt-hours of power available if used at the ½-amp discharge rate. (This capacity is reduced for higher discharge rates). The full charge capacity of the battery system is 3200 watt-hours at the ½-amp discharge rate. With the design modifications necessary to meet the electrical performance, the spacecraft will require about 4900 watt-hours to survive the lunar night.

D. *Planar Array Alignment*

As currently designed, the planar array is pointed at Earth by using the solar panel as a reference. This scheme would of necessity have to be changed for landing at night. The planar array could be oriented through the use of a spacecraft "plumb bob" in connection with trajectory data.

E. *Other Experiments*

With sufficient power, the micrometeorite and soil mechanics experiments could be operated during the lunar night. However, higher mechanical friction and decreased electrical resistance will be experienced at the low temperatures, so any equipment operated at night will require perhaps four times as much power as when operated during the day. Extensive mechanical redesign would be required to enable these equipments to operate during lunar night.

F. *Reliability*

Assuming ample power is available, the probability of conducting successful TV operations after surviving a lunar night is about 0.1.

G. *Flash Pictures*

There exists today lightweight, low-power-consuming, reliable flash equipment which may enable the spacecraft to take pictures at night to perhaps a distance of 20 feet.

H. *Image Orthicon*

The image orthicon is technically feasible for taking night pictures. In a recent demonstration at Hughes Aircraft Company, an image-orthicon system took good quality pictures in a darkened projection room with only a movie projector pilot light for illumination.

I. *Flat Lighting*

The flat lighting characteristic of Earth shine would reduce picture contrast.

J. *Limited Night-Time Landing*

If no landing is attempted from 50 hr before day/night terminator until lunar midnight, and the landing area can be selected between 8 and 65 deg W longitude, and a minimum daily window of 30 minutes is required, and a true anomaly from −6 to +12 deg is assumed; the number of launch days shown in Table H-1 will result. For this launch situation, the spacecraft would have a probability of 0.7 of surviving the lunar night and taking TV pictures the following day. The third column of Table H-1 is for the condition where 100 hr of possible landing daylight is available.

Table H-1. Launch opportunities

Year	Month	Number of launch days	
		Daylight (100 hr)	Night landing
1964	July	5	5
	Aug	5	5
	Sep	5	5
	Oct	2	5
	Nov	0	6
	Dec	0	5
1965	Jan	0	3
	Feb	0	2
	Mar	3	4
	Apr	4	6
	May	6	6
	June	6	6
	July	6	6
	Aug	6	6
	Sep	6	6
	Oct	3	6
	Nov	1	6
	Dec	0	6

Assumptions:
1. No landing between 50 hr before day/night terminator and lunar midnight.
2. Period: Mid-1964 through 1965.
3. Landing region: maria, 8 to 65 deg W longitude.
4. Minimum daily window: 30 minutes.
5. True anomaly: −6 to +12 deg.

APPENDIX I

Typical Mission Profile

ASSUMPTIONS

1. Touchdown within first two hours of Goldstone visibility, with a minimum of 70 hr until day/night terminator.

2. Diffractometer and alpha scattering experiments can operate simultaneously.

3. Micrometeorite experiment operates at all times, except during TV operation.

4. Only the TV requires the high power transmitter; and the transmitter requires one hour cooling after each one hour continuous duty cycle.

5. Operating temperature constraints:
   a. Surface sampler and soil processor will not be operable within 32 hr of day/night terminator.
   b. The television system will not be operable within ten hours of day/night terminator.

6. The Sample Transport System is assumed to be capable of redesign to accommodate the particular sample sequence in the attached sequence.

7. Alpha scattering experiment requires a standard and background sample on either side of the lunar analysis, with no more than a six-hour gap between any of these three.

OPERATIONAL SEQUENCE (cf Fig.I-1)

1. Approach-descent television pictures (camera #4).
2. Wide angle and narrow angle TV (½ hour each).
3. Soil mechanics experiment (½ hour) with TV snapshots.
4. Surface sampler operation (½ hour) with acquisition of first lunar sample.
5. Background count for diffractometer. Turn on micrometeorite detector and operate continuously, except during television operation.
6. Standard samples for diffractometer and alpha scattering.
7. Pulverize and feed first lunar sample to diffractometer, acquire two more lunar samples with surface sampler and TV, process and analyze with alpha scattering.
8. Rescan first lunar sample with diffractometer (can be received overseas).
9. Feed standard samples to alpha scattering and perform background count overseas.
10. Re-run alpha scattering background overseas and read out at start of second Goldstone visibility.
11. Perform ½ hour each of wide angle and narrow angle TV.
12. Acquire and process three lunar samples with TV coverage; feed to diffractometer and alpha scattering and analyze.
13. Feed standards to alpha scattering and analyze.
14. Acquire three more lunar samples with TV coverage, pulverize and analyze with diffractometer and alpha scattering.

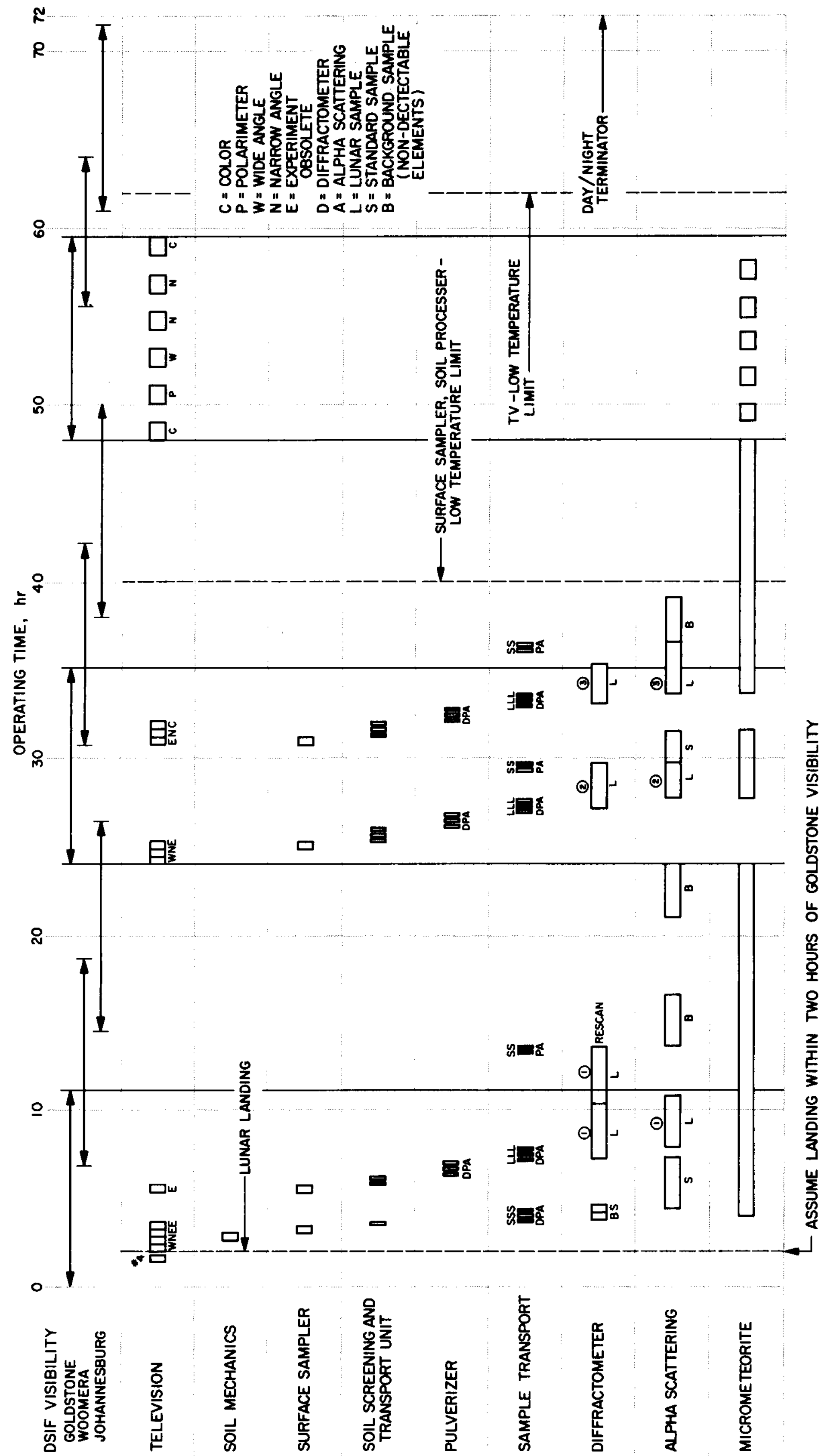

Fig. I-1. Surface operations

15. Feed standards to alpha scattering and analyze.
16. Pass low temperature operational limit of surface sampler and soil processing system 32 hours before terminator.
17. Perform alternate one-hour TV and micrometeorite experiments, allowing high-power transmitter to cool between cycles.
18. Pass low temperature operational limit of TV system ten hours before terminator.
19. Day/night terminator.

APPENDIX J

Direct-Ascent Advantage to Launch Vehicle

Utilization of a direct-ascent trajectory eliminates those *Centaur* design features which are necessary for it to execute a parking orbit. Included in this category are:

1. **Elimination of the *Centaur* Sun-Seeker Requirement.** When in a parking orbit, the *Centaur* must achieve and maintain an attitude which points its engine nozzles at the Sun to minimize H_2 boil-off. The elimination of this requirement permits the use of a three- (instead of four) gimbal guidance platform, and may permit the use of an alternative to the inertial-platform guidance system. This requirement can also be eliminated by reducing the time in parking orbit from the 34-minute maximum to perhaps 20 minutes.

2. **Elimination of the *Centaur* Restart Requirement.** This would reduce H_2O_2 system complexity since the ullage rockets would not be required for restart. The programmer would be simplified. The propellant required to chill down the engines at restart would be saved. A simpler H_2 vent system could be used, since no zero-g valving would be necessary. The uncertainty regarding propellant behavior under zero-g conditions would be eliminated.

3. **Reduction in Launch Vehicle Guidance Requirements.** Launch-to-injection time is shorter when the time in parking orbit is reduced, so larger gyro drifts could be tolerated. The direct-ascent case is equivalent to zero parking orbits in this instance.

4. **Possible Reduction in *Centaur* Development Time** results from 1, 2, and 3 above. It is unlikely that any significant decrease in *Centaur* vehicle weight will result from these advantages; perhaps 50 to 75 pounds can be saved without redesign.

5. **Simpler Telemetry and Tracking.** The vehicle, from launch through injection, is always within sight of the Cape or Bermuda for direct-ascent trajectories. Again this same situation can be approximated by limiting parking-orbit duration.